

Thesis  
F2



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DESIGN LOADS FOR HORIZONTAL TAIL SURFACES  
FOR AIRPLANES.

By  
*to ter*  
Lieutenant Delmer S. Fahrney, U.S.Navy, 1898-  
and  
Lieutenant Ward C. Gilbert, U.S.Navy., 1898-

Submitted in partial fulfillment of the  
requirements for the degree of Master of  
Science from the Massachusetts Institute of  
Technology.

Authors:

Approved by:

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For the Department of  
Aeronautical Engineering.

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## OBJECT:

The object of this report is to present a simple, rational method for determining the value of design loads to be used for airplane horizontal tail surfaces and fuselage tail structures.

## INTRODUCTION:

Investigation establishes the fact that the design loads sought, must be determined from two distinct conditions of flight. The basic assumptions made are:

(1) That the dive condition at terminal velocity, or at some limiting velocity, depending upon the type and service expected of the aircraft, will design the front spar of the horizontal stabilizer.

(2) That the abrupt pull-up condition from some limiting velocity, depending upon the type and service expected of the aircraft, will design the rear spar of the horizontal stabilizer and the elevator.

These assumptions are conclusively borne out by an examination of the pressure distribution tests conducted by the N.A.C.A., upon the tail surfaces of the F6C-4 and PW-9 airplanes.

More specifically then, to arrive at a successful accomplishment of this investigation, the following determinations must be made:

(1) A reasonably simple and rational expression that will give the total normal tail load upon the horizontal surfaces at the particular limiting diving speed specified for the type.

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These assumptions are conclusively borne out by an examination of the pressure distribution tests conducted by the N.A.C.A., upon the tail surfaces of the T6C-4 and PW-3 airplanes.

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(1) A reasonably simple and rational expression that will give the total normal tail load upon the horizontal surfaces at the particular limiting diving speed specified for the type.

(2) A distribution of the normal tail load found in (1), in such a manner as to impose a critical loading on the front spar of the horizontal stabilizer.

(3) A reasonably simple and rational expression, that will give the total normal tail load upon the horizontal surfaces for the abrupt pull-up condition, at a particular limiting diving speed for the type, from which the abrupt pull-up shall be executed.

(4) A distribution of the total normal tail load found in (3), in such a manner as to impose a critical loading upon the rear spar of the horizontal stabilizer.

A special effort has been made in the compilation of this report, to include all pertinent calculations and references, in order that the results attained may be substantiated by the material to be found within its covers. These data and comparative calculations are contained in the " APPENDIX".

#### DISCUSSION:

Dive Condition: For the purpose of determining the total normal tail load, calculations were first made upon the F6C-4 airplane by the method set forth in A.D.M. 1061. A limited amount of basic data necessary to the determination of the normal tail load were available. In point, stagger, relative wing efficiency, and equivalent monoplane aspect ratio had to be determined. The calculation of the tail load was, therefore, extended somewhat for this airplane. But granting this extension, the impression received was to the effect that the determination of the normal

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tail load by the method of A.D.M. 1061, was exceedingly laborious. And having in mind the specific purpose of this investigation, to wit, - to attain the desired results by simple and rational methods, the investigation naturally proceeded to the study of means by which such simplification could be brought about.

The N.A.C.A. Report No. 307 "Pressure Distribution over Tail Surfaces of F6C-4 Airplane", specified neither the test weight of the airplane, nor the C.G. location. In order to make a precise comparison between calculated tail loads and loads obtained as the result of actual flight tests, the tail load calculations for the F6C-4 airplane were repeated. The gross weight and C.G. location for the first set of calculations, were assumed to be those corresponding to the fully loaded condition. In an indirect manner later information was received, which indicated that an airplane with a gross weight and C.G. location different from that of the fully loaded airplane, had been used in the N.A.C.A. tests, and these values were employed in the second set of calculations on the F6C-4.

It was the comparison of the results obtained in these two sets of calculations, that shed light upon a possible simplification of the tail load formula given in A.D.M. 1061. The results indicated that near 'zero lift' for the airplane, the values of  $K_M$  c.g. agreed almost absolutely, for the two C.G. locations.

Comparative calculations of  $K_M$  c.g. were then made for extreme C.G. locations, and the results plotted in Fig. 22. The curves showed that very wide ranges of C.G. location, effected but slight

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changes in the value of  $K_M$  c.g. And for the usual range of C.G. location employed in actual design, the change in  $K_M$  c.g. would be much smaller. It was then decided that the airfoil moment coefficient at zero lift, might well be employed in the tail load formula, instead of  $K_M$  c.g. , which is determined only after lengthy calculations. The agreement found to exist, by reason of using the value of  $K_M$  of the airfoil at zero lift, and of using  $K_M$  c.g. for the airplane, upon the resulting tail load, was pronounced "good".

For the F6C-4 airplane, calculations by the method of A.D.M. 1061, gave a tail load of 925 lbs. at 247 m.p.h., as compared with 468 lbs. at the same speed recorded by actual pressure tests by the N.A.C.A. The calculations give an average tail load of 28 lbs. per sq.ft., while the average tail load by actual flight test is but 14 lbs. per sq.ft. The calculated normal tail load for the dive condition at terminal velocity of 286 m.p.h., would be 1240 lbs., or 38 lbs. per sq.ft. The normal tail load at 286 m.p.h. determined on the basis of actual flight test at 247 m.p.h., would be 630 lbs., or an average loading of but 19 lbs. per sq. ft. Since this load would correspond to a design condition, it appears that either 50% of the calculated normal tail load is borne by the fuselage, or that pressure tests recorded low readings. It can hardly be disputed that 38 lbs. per sq.ft. is a more reasonable value for design load, than is 19 lbs. per sq.ft. for this type of aircraft.

It is unquestionably true that a considerable part of the total balancing load on the tail will be borne by the fuselage. It does not seem likely that as much as 50% would be so borne, however.

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The form of the particular fuselage aft of the C.G. would be a governing factor. At this point the conclusion must be drawn, to the effect that the actual normal tail load borne by the horizontal tail area, is at present an indeterminate quantity, and will remain as such, until a vast amount of pressure distribution data, covering both fuselage and tail surfaces on a large number of airplanes, is at hand. The calculated normal tail loads, therefore, will always be approximate and conservative in varying degrees, on various airplanes. Then too, the results obtained cannot possess greater precision than the basic airfoil data, and these are not in close agreement. A.D.M. 1061 gives a terminal velocity of 312 m.p.h. for the PW-9 airplane. For this same airplane, Gottingen tunnel data, which formed the basis for PW-9 computations, by the method of A.D.M. 1061, contained in this report, gives a terminal velocity of 330 m.p.h. The maximum normal tail load as computed in A.D.M. 1061 is 1480 lbs.; this difference in terminal velocity alone, would result in an increase in tail load to 1660 lbs. It appears, therefore, that any attempt to calculate with precision, the actual normal tail load upon the horizontal tail surfaces alone, is of academic interest only, and a fruitful source of wasted time and effort. However, if with a reasonable expenditure of time and effort, a value of the design normal tail load can be calculated, which will form a basis for tail surface design, it might possess some virtue. A method for determining the value of the design normal tail load by an approximation to the method indicated in A.D.M. 1061, has been evolved in

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this report; and so far as comparative calculations have proceeded, reasonable results have been secured.

The expression for normal tail load set forth in A.D.M. 1061 is as follows:

$$\text{Normal Tail Load} = K_M \text{ c.g.} \times A_w \times \frac{C}{d} \times V^2$$

It is proposed that  $K_M \text{ c.g.}$  be replaced by airfoil  $K_M$  at zero lift, and that  $V$  be equal to the maximum limiting diving speed for the particular type of aircraft under consideration. For the pursuit or fighting, attack and training types of aircraft,  $V$  will be the terminal velocity of the airplane as given from the following formula:

$$V_t = \sqrt{\frac{V_{\max}^3 \times W}{375 \times \eta_{\max} \times \text{B.H.P.}}}$$

The origin of this formula is unknown. Values of terminal velocity for various types of aircraft have been computed by the above formula, and also by the drag formula,  $W \cos 90^\circ = K_x A_w V_t^2$

Phenomenally close agreement of values of terminal velocity, calculated from both these formulas was secured, and the comparative results have been set forth in the "APPENDIX" to this report on page 75.

The employment of the power absorbed formula for terminal velocity, obviates the necessity of first determining airplane  $K_x$ .

For the purpose of calculating design normal tail loads, for other than the single seater fighter, attack and training types, limiting velocities have been proposed, and have been set forth in this report in "RESULTS".

The use of the zero lift airfoil moment coefficient, in the formula for normal tail load, with velocities less than terminal, is an approximation; but since at very high speeds, great changes in speed are

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accompanied by slight changes in angle of attack, which are accompanied by corresponding small changes in  $K_M$  e.g. . this approximation can be justified.

The distribution of the normal tail load was worked out from N.A.C.A. pressure distribution data on the P6C-4 and PW-9 airplanes, and an expression for design load in lbs. per inch of run, for the front spar of the horizontal stabilizer has been evolved. These data will be found in this report under "RESULTS".

Pull-up Condition: The Rhode formula for the pull-up condition has been examined, but since its application in its present state of development, is limited to the fighting type of aircraft, another viewpoint was assumed. It appeared that the most severe loading on the horizontal tail surfaces for the pull-up condition, would be the arithmetic sum of the normal tail load that existed by reason of the dive condition, at the instant movement of elevator was begun, and the normal load imposed on the rear spar, by reason of the inclination of the elevator at an angle of  $25^\circ$ . This assumes no loss of velocity and no movement of the tail to a new angle of attack. So much of the tail load as existed by reason of the dive alone, is calculated in a manner similar to that described in the foregoing discussion on design load for front spar. For so much of the additional load imposed by reason of the inclination of the elevator, it was assumed that the elevator is a flat plate inclined at an angle of  $25^\circ$  to the stabilizer. The viewpoint herein taken, offers the simplest possible method of attack to the problem of pull-up



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The distribution of the normal tail load was worked out from N.A.C.A. pressure distribution data on the F5C-4 and PW-9 airplanes, and an expression for design load in lbs. per inch of run, for the front spar of the horizontal stabilizer has been evolved. These data will be found in this report under "RESULTS".

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loads, and so far as the investigation has proceeded, gives good results. It will probably be received as unscientific, but seems to retaliate with the virtues of simplicity and good results. Such a formula permits of the introduction of the two most essential variables for the pull-up condition, viz., the elevator area and the speed at which the maneuver is executed. It only remains then, to establish limiting abrupt pull-up velocities, depending upon the type of aircraft and the type of flying to which the aircraft may be expected to be subjected. This has been done, and these limiting velocities appear under the caption "RESULTS", in this report.

These proposed limiting velocities were plotted against lbs. per H.P. for a large number of airplanes, and it is interesting to note that distinct types grouped themselves together, along a continuous contour, with the exception of the training plane group. It is felt that a reasonable limiting velocity for that type should be about 160 m.p.h. It appears that limiting velocity for abrupt pull-up, is a function of lbs. per H.P.

The distribution of the tail load for the pull-up condition was made in accordance with N.A.C.A. pressure distribution data on the F6C-4 and PW-9 airplanes. Expressions for loads in lbs. per inch of run for both the elevator spar and the stabilizer rear spar have been evolved, and appear in this report on Fig. 2 . This is as far as this analysis could be carried, since the running load on elevator is supported by the elevator hinges, which in turn, impose concentrated loads upon the rear spar of the horizontal stabilizer.

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The distribution of the tail load for the pull-up condition was made in accordance with W.A.C.A. pressure distribution data on the 76C-4 and PW-9 airplanes. Expressions for loads in lbs. per inch of run for both the elevator spar and the stabilizer rear spar have been evolved, and appear in this report on Fig. . This is as far as this analysis could be carried, since the running load on elevator is supported by the elevator hinges, which in turn, impose concentrated loads upon the rear spar of the horizontal stabilizer.



Referring to Fig. 2 , it would be more conservative to consider the load on the leading edge of the stabilizer equal to "w" , instead of "2w" , since such a distribution would impose a more severe load on the rear spar. However, the distribution made in Fig. 2 is in accord with the actual distribution, given in the N.A.C.A. Reports for the tail surfaces of the F6C-4 and PW-9 airplanes.

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RESULTS.

RESULTS.

# DESIGN NORMAL TAIL LOADS FOR FRONT SPAR OF HORIZONTAL STABILIZER:

## Pursuit or Fighting Airplanes:

Design normal tail load is that imposed by terminal velocity condition.

$$\text{Normal Tail Load} = K_{M_0} \times A_w \times \frac{C}{d} \times V_t^2$$

where,  $K_{M_0}$  = Airfoil  $K_M$  at zero lift.

$A_w$  = Wing area.

$C$  = M.A.C.

$d$  = Distance from C.G. To rudder post.

$$V_t = \sqrt{\frac{V_{\max}^2 \times W}{375 \times \gamma_{\max} \times \text{B.H.P.}}}$$

## Attack Airplanes:

(As above)

## Training Airplanes:

(As above)

## Cargo and Bombardment Airplanes:

Design normal tail load is that imposed at a limiting diving speed, which is determined by the character of flying required of the type. The proposed limiting diving speed for these types is 200 m.p.h.

$$\text{Normal Tail Load} = K_{M_0} \times A_w \times \frac{C}{d} \times V^2$$

where,  $V$  = Limiting diving speed, which is in turn equal to maximum allowable diving speed, plus a reasonable margin of safety. A 50% margin of safety is suggested, thereby giving an allowable diving speed of 163 m.p.h.

Limiting diving speeds established at this time, should probably be increased from time to time, as power is stepped up.

# DESIGN NORMAL TAIL LOADS FOR FRONT SEAT OF HORIZONTAL

## STABILIZER:

### Parasit or Fighting Airplanes:

Design normal tail load is that imposed by terminal velocity condition.

Normal Tail Load  $K_N \times A_W \times \frac{C}{D} \times V^2$   
where,  $K_N$  Airfoil  $K_N$  at zero lift.

$A_W$  - Wing area.

$C$  - M.A.C.

$d$  Distance from C.G. to rudder post.

$$V^2 = \frac{V_{max}^2 \times W}{375 \times \max \times B.H.P.}$$

### Attack Airplanes:

(As above)

### Training Airplanes:

(As above)

### Cargo and Bombardment Airplanes:

Design normal tail load is that imposed at a limiting diving speed, which is determined by the character of flying required of the type. The proposed limiting diving speed for these types is 200 m.p.h.

$$\text{Normal Tail Load } K_N \times A_W \times \frac{C}{D} \times V^2$$

where,  $V$  Limiting diving speed, which is in turn equal to maximum allowable diving speed, plus a reasonable margin of safety. A 50% margin of safety is suggested, thereby giving an allowable diving speed of 150 m.p.h.

Limiting diving speeds established at this time, should properly be increased from time to time, as power is stepped up.

DESIGN TAIL LOAD FOR ABRUPT PULL-UP CONDITION:

Dynamic Tail Load = Normal tail load at limiting pull-up velocity for dive condition + down load due to inclination of elevator, considered as a flat plate inclined at an angle of  $25^\circ$ .

$$\text{Dynamic Tail Load} = (K_{M_o} \times A_w \times \frac{C}{d} \times V_{lim}^2) + (.00202 \times A_e \times V_{lim}^2)$$

Derivation of formula:

The symbols appearing in so much of the formula as determines the normal tail load due to the dive, are the same as have been described heretofore, for the dive condition.  $V_{lim}$  is the design limiting velocity, which in turn is equal to the allowable abrupt pull-up velocity, plus a reasonable margin of safety. The assignment of  $V_{lim}$  for any particular type, will depend upon the service expected of the airplane.  $A_e$  = elevator area in sq.ft.

$$P_{90^\circ} = .0032 \times A_e \times V_{lim}^2$$

$$\frac{P_{25^\circ}}{P_{90^\circ}} = .7$$

$$P_{25^\circ} = .7 \times .0032 \times A_e \times V_{lim}^2$$

Down load due to inclination of elevator  $25^\circ$  =

$$.7 \times .0032 \times A_e \times V_{lim}^2 \times \cos 25^\circ = .00202 \times A_e \times V_{lim}^2$$

Examples:F6C-4 Airplane:

For this type, limiting abrupt pull-up velocity assumed to be 190 m.p.h.

$$\begin{aligned} \text{Dynamic Tail Load} &= (.00021 \times \frac{58.7}{167.3} \times 252 \times 190^2) + (.00202 \times 14.8 \times 190^2) \\ &= (669 + 1080) = 1749 \text{ lbs.} \end{aligned}$$

$$\text{Average loading} = \frac{1749}{32.9} = 53.1^*/\text{sq.ft.}$$

# DESIGN TAIL LOAD FOR ABRUPT PULL-UP CONDITION:

Dynamic Tail Load Normal tail load at limiting pull-up velocity for dive condition - down load due to inclination of elevator, considered as a flat plate inclined at an angle of 25°.

$$\text{Dynamic Tail Load } (K_M \times A_w \times \frac{C}{D} \times V_{lim}^2) - (.00205 \times A_e \times V_{lim}^2)$$

Derivation of formula:

The symbols appearing in so much of the formula as determines the normal tail load due to the dive, are the same as have been described heretofore, for the dive condition.  $V_{lim}$  is the design limiting velocity, which in turn is equal to the allowable abrupt pull-up velocity, plus a reasonable margin of safety. The assignment of  $V_{lim}$  for any particular type, will depend upon the service expected of the airplane.  $A_e$  elevator area in sq.ft.

$$P_{25} = .0025 \times A_e \times V_{lim}^2$$

$$\frac{P_{25}}{P_{90}} = .7$$

$$P_{25} = .7 \times .0025 \times A_e \times V_{lim}^2$$

Down load due to inclination of elevator 25°

$$.7 \times .0025 \times A_e \times V_{lim}^2 \cos 25^\circ .00205 \times A_e \times V_{lim}^2$$

Examples:

Vec-4 Airplane:

For this type, limiting abrupt pull-up velocity assumed to be 120 m.p.h.

$$\text{Dynamic Tail Load } (.00251 \times \frac{58.7}{16.7} \times 252 \times 120^2) - (.00205 \times 14.8 \times 120^2)$$

$$(622 - 1080) = 1742 \text{ lbs.}$$

$$\text{Average loading } \frac{1742}{32.2} = 53.1 \text{ g.p.f.t.}$$



DESIGN TAIL LOADS FOR ABRUPT PULL-UP CONDITION:PW-9 Airplane:

As in the case of the F6C-4 calculation, limiting abrupt pull-up velocity assumed to be 190 m.p.h.

$$\begin{aligned}\text{Dynamic Tail Load} &= (.00020 \times \frac{59.15}{177.3} \times 240.76 \times 190^2) + (.00202 \times 9.9 \times 190^2) \\ &= (580 + 722) = 1302 \text{ lbs.}\end{aligned}$$

$$\text{Average loading} = \frac{1302}{30.3} = 43.0 \text{ \# / sq.ft.}$$

A-3 Airplane:

For this type, limiting abrupt pull-up velocity assumed to be 160 m.p.h.

$$\begin{aligned}\text{Dynamic Tail Load} &= (.00021 \times \frac{64.3}{200} \times 353 \times 160^2) + (.00202 \times 19.91 \times 160^2) \\ &= (610 + 1030) = 1640 \text{ lbs.}\end{aligned}$$

$$\text{Average loading} = \frac{1640}{47.31} = 34.7 \text{ \# / sq.ft.}$$

B-2 Airplane:

For this type, limiting pull-up velocity assumed to be 120 m.p.h.

$$\begin{aligned}\text{Dynamic Tail Load} &= (.000246 \times \frac{108.8}{377} \times 120^2) + (.00202 \times 49.6 \times 120^2) \\ &= (1532 + 1442) = 2974 \text{ lbs.}\end{aligned}$$

$$\text{Average loading} = \frac{2974}{146.4} = 20.3 \text{ lbs. / sq.ft.}$$

LB-7 Airplane:

For this type, limiting abrupt pull-up velocity assumed to be 120 m.p.h.

$$\begin{aligned}\text{Dynamic Tail Load} &= (.000228 \times \frac{96.2}{340} \times 1150 \times 120^2) + (.00202 \times 61.2 \times 120^2) \\ &= (1070 + 1780) = 2850 \text{ lbs.}\end{aligned}$$

$$\text{Average loading} = \frac{2850}{144.6} = 19.7 \text{ lbs. / sq. ft.}$$

# DESIGN TAIL LOADS FOR ABRUPT BUILD-UP CONDITION:

## PW-3 Airplane:

As in the case of the T6C-4 calculation, limiting abrupt build-up velocity assumed to be 120 m.p.h.

$$\text{Dynamic Tail Load} = (.00020 \times \frac{52.12}{177.3} \times 240.76 \times 120) - (.00202 \times 2.2 \times 120) \\ (580 - 722) \quad 1302 \text{ lbs.}$$

$$\text{Average loading} = \frac{1302}{30.3} = 43.0 \text{ \# / sq. ft.}$$

## A-3 Airplane:

For this type, limiting abrupt build-up velocity assumed to be 160 m.p.h.

$$\text{Dynamic Tail Load} = (.00021 \times \frac{64.3}{300} \times 323 \times 160) - (.00202 \times 12.2 \times 160) \\ (610 - 1030) \quad 1640 \text{ lbs.}$$

$$\text{Average loading} = \frac{1640}{47.31} = 34.7 \text{ \# / sq. ft.}$$

## B-3 Airplane:

For this type, limiting abrupt build-up velocity assumed to be 120 m.p.h.

$$\text{Dynamic Tail Load} = (.000246 \times \frac{108.8}{377} \times 120) - (.00202 \times 42.6 \times 120) \\ (1222 - 1442) \quad 224 \text{ lbs.}$$

$$\text{Average loading} = \frac{224}{146.4} = 1.5 \text{ \# / sq. ft.}$$

## B-7 Airplane:

For this type, limiting abrupt build-up velocity assumed to be 120 m.p.h.

$$\text{Dynamic Tail Load} = (.000228 \times \frac{26.5}{340} \times 1120 \times 120) - (.00202 \times 61.2 \times 120) \\ (1070 - 1780) \quad 2850 \text{ lbs.}$$

$$\text{Average loading} = \frac{2850}{144.6} = 19.7 \text{ \# / sq. ft.}$$

DESIGN TAIL LOADS FOR ABRUPT PULL-UP CONDITION:PT-3A Airplane:

For this type, limiting abrupt pull-up velocity assumed to be 160 m.p.h.

$$\begin{aligned} \text{Dynamic Tail Load} &= (.00021 \times \frac{56}{222} \times 300 \times 160^2) + (.00202 \times 16.9 \times 160) \\ &= (407 + 875) = 1282 \text{ lbs.} \end{aligned}$$

$$\text{Average loading} = \frac{1282}{33.6} = 38.1 \text{ lbs. / sq.ft.}$$

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DESIGN TAIL LOADS FOR ABRUPT FULL-UP CONDITION:

Pt-3A Airplane:

For this type, limiting abrupt pull-up velocity assumed to be 160 m.p.h.

Dynamic Tail Load  $(.00051 \times \frac{50}{225} \times 300 \times 160) - (.00205 \times 16.9 \times 160)$

(407 - 875) 1588 lbs.

Average loading  $\frac{1588}{33.6} = 38.1 \text{ lbs. / sq.ft.}$

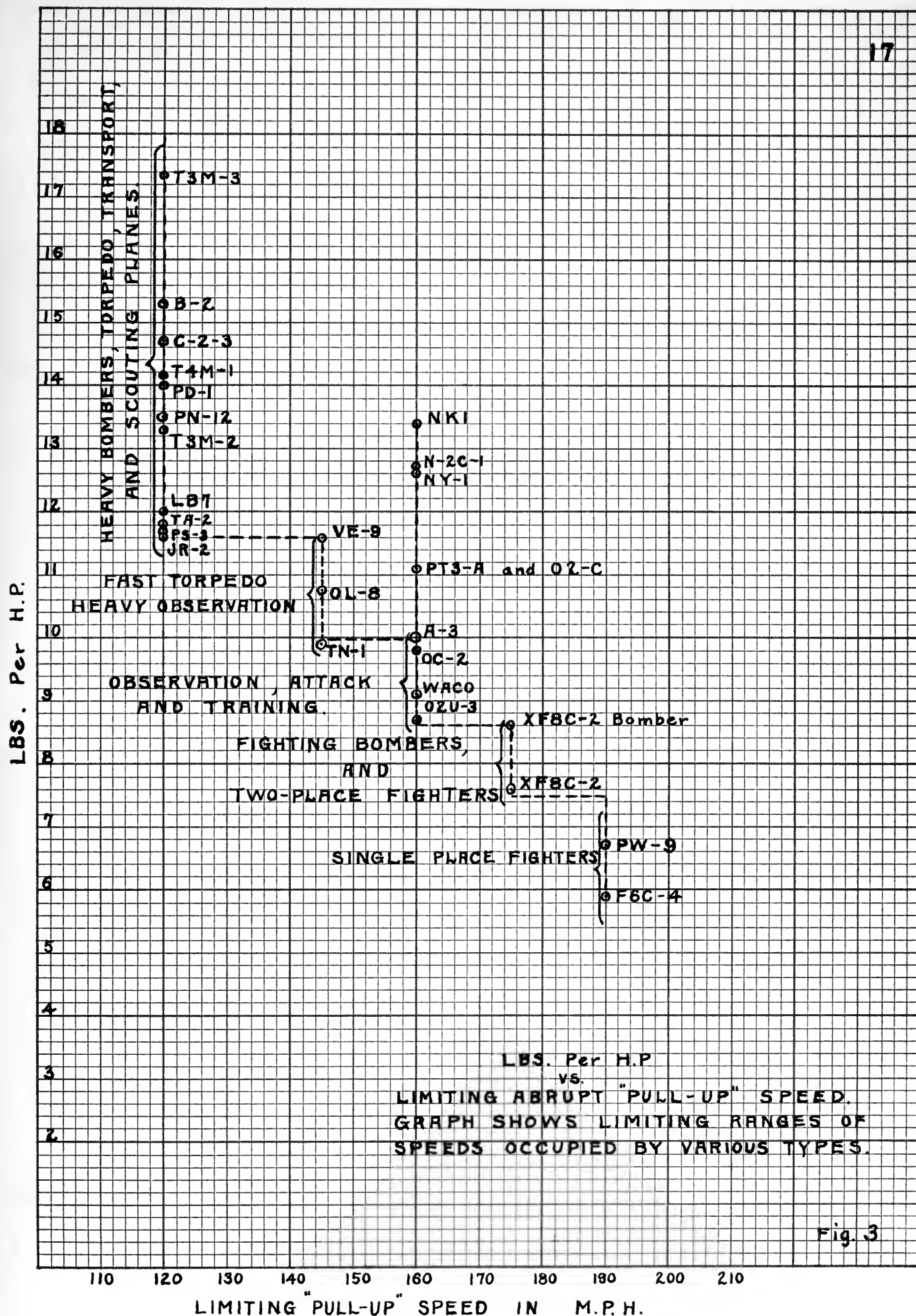


Fig. 3



The following remarks have been copied directly from A.D.M.1061 :

In order to further insure that the construction of the tail surfaces shall never be too flimsy to withstand handling, fabric tension and to resist flutter at high speed, tail surfaces shall never be designed for an average loading less than the values given in the following table:

Pursuit	- 50 lbs. / sq.ft.
Attack and Observation	- 40 lbs. / sq.ft.
Cargo and Bombardment	- 25 lbs. / sq.ft.
Training	- 40 lbs. / sq.ft.

The following remarks have been copied directly from A.D.N.1061 :

In order to further insure that the construction of the tail surfaces shall never be too flimsy to withstand handling, fabric tension and to resist flutter at high speed, tail surfaces shall never be designed for an average loading less than the values

given in the following table:

Training	- 40 lbs. / sq.ft.
Cargo and Bombardment	- 55 lbs. / sq.ft.
Attack and Observation	- 40 lbs. / sq.ft.
Turnout	- 50 lbs. / sq.ft.



# BIBLIOGRAPHY :

A.D.M. 1061 - "Proposed Method of Determining Design Tail Loads for Airplanes".

A.D.M. 900 - "The Lift Distribution in any Biplane"

A.D.M. reports are Air Corps Technical Reports, prepared by Air Corps Materiel Division, Wright Field, Dayton, Ohio.

N.A.C.A. Report No. 233 - "The Aerodynamic Characteristics of seven frequently used wing sections at full Reynolds Number.

N.A.C.A. Report No. 307 - "The Pressure Distribution Over the Horizontal and Vertical Tail Surfaces of the F6C-4 Pursuit Airplane in Violent Maneuvers".

N.A.C.A. Report No. 331 - "Collection of Wind Tunnel Data on Commonly Used Wing Sections"

N.A.C.A. Advance Report - "Tail Surface Loads and Pressures on PW-9 Airplane".

N.A.C.A. reports are prepared by National Advisory Committee for Aeronautics, Washington, D.C.

A.C.I.C. 607 - "Induced Drag of any Biplane".

A.C.I.C. 629 - "Determination of Structural Airplane Drag".

A.C.I.C. reports are Air Corps Information Circulars, published by Chief of Air Corps, Washington, D.C.

Handbook of Instructions for Airplane Designers, prepared by U.S. Army Materiel Division, Wright Field, Dayton, Ohio.

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N.A.C.A. Report No. 233 - "The Aerodynamic Characteristics of seven frequently used wing sections at full Reynolds Number.

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Handbook of Instructions for Airplane Designers, prepared by U.S. Army Materiel Division, Wright Field, Dayton, Ohio.

APPENDIX:

**APPENDIX:**

CURTISS F6C-4 FIGHTING AIRPLANE

TAIL LOAD COMPUTATIONS.

CURTIS P-40 FIGHTING AIRPLANE  
TAIL LOAD COMPUTATIONS.

BASIC DATA ON F6C-4 AIRPLANE:

Gross Weight (W) -	2796
Wing Section -	Clark "Y"
M.A.C. (C) -	58.7"
Span, upper -	31.5'
Span, lower -	26.0'
Chord, upper, (average) -	62.6"
Chord, lower, (average) -	49.4"
Gap -	53.31"
Stagger, at L.E., at fuselage -	38.5"
C.G. % M.A.C. -	34.1%
C.G. % M.A.C. below -	35.75%
Area of horizontal tail surfaces -	32.9 sq.ft.
Total wing area -	252 sq.ft.
Distance from C.G. to tail post - (d)	167.5"
Area, upper wing -	158 sq.ft.
Area, lower wing -	94 sq.ft.
Maximum speed, sea level -	157 m.p.h.
B.H.P. 435 at 1950 R.P.M. -	
Diameter of propeller -	8.665 ft.
L.E.M.A.C. 23.6" ahead L.E.L.W.	

Authority: Curtiss Aeroplane & Motor Company, Inc.



# BASIC DATA ON REC-4 AIRPLANE:

8.665 ft.		I.E.M.A.C. 23.6" ahead H.E.L.W.
		Diameter of propeller -
		B.H.P. 435 at 1250 R.P.M. -
		Maximum speed, sea level -
127 m.p.h.		Area, lower wing -
94 sq.ft.		Area, upper wing -
158 sq.ft.		Distance from C.G. to tail post - (d)
167.5"		Total wing area -
252 sq.ft.		Area of horizontal tail surfaces -
32.9 sq.ft.		C.G. M.A.C. below -
35.75"		C.G. M.A.C. -
34.15"		Stagger, at I.E., at fuselage -
38.5"		Gap -
53.31"		Chord, lower, (average) -
49.4"		Chord, upper, (average) -
62.6"		Span, lower -
56.0'		Span, upper -
31.5'		M.A.C. (C) -
58.7"		Wing section -
		Gross Weight (W) -
2796	Clark "Y"	

Authority: Curtiss Aeroplane & Motor Company, Inc.

CLARK "Y" CHARACTERISTICS:

$\alpha$	$C_L$	$C_D$	$C_m$ 25%	$K_y$	$K_x$	$K_m$ 25%
-6.0	-.060	.0108	-.083	-.0001536	.00002765	-.0002125
-4.5	.045	.0107	-.080	.0001152	.0000274	-.000205
-3.0	.167	.0119	-.078	.0004275	.00003045	-.0001997
-1.5	.268	.0139	-.078	.0006860	.0000356	-.0001997
0	.384	.0172	-.070	.0009830	.0000440	-.0001792
1.5	.501	.0228	-.059	.0012820	.00005835	-.000151
3.0	.602	.0288	-.073	.001540	.0000737	-.000187
6.0	.819	.0464	-.079	.002098	.000119	-.0002022
9.0	1.034	.0700	-.054	.00265	.000179	-.0001383
12.0	1.231	.0985	-.036	.00315	.000252	-.0000922
15.0	1.367	.1272	-.077	.00350	.0003255	-.000197
18.0	1.283	.2108	-.056	.003283	.000540	-.0001434
21.0	1.081	.2946	-.048	.00277	.000754	-.000123

Authority; N.A.C.A. Report No. 233. (High Density Tunnel Data).

# CLARK "Y" CHARACTERISTICS:

C	C	Cm S24 K	K	Km S24
21.0	1.081	.246	-.048	.0057
18.0	1.283	.2108	-.056	.003283
15.0	1.367	1.273	-.077	.00320
12.0	1.331	.0282	-.036	.00312
9.0	1.034	.0700	-.054	.00262
6.0	.812	.0464	-.072	.002028
3.0	.602	.0288	-.073	.001240
1.2	.201	.0228	-.052	.0012820
0	.384	.0172	-.070	.0002830
-1.2	.268	.0132	-.078	.0006860
-3.0	.167	.012	-.078	.0004272
-4.2	.042	.0107	-.080	.000122
-6.0	-.060	.0108	-.082	-.0001226

Authority: N.A.C.A. Report No. 533. (High Density Tunnel Data).

DETERMINATION OF CORRECTED  $\frac{L}{D}$  FOR CELLULE:

$$\text{Corrected } \frac{L}{D} \text{ cellule} = \frac{1}{\frac{D}{L} \text{ model} + B \times K_y}$$

$$B = 125 \left( \frac{C_l^2 + C_u^2 + 2s C_l C_u}{A} - \frac{1}{AR \text{ model}} \right)$$

$$A = (\text{total wing area} + \text{area under fuselage}) = 264.6 \text{ sq.ft.}$$

$$B = 125 \left( \frac{4.115^2 + 5.22^2 + 2 \times .53 \times 4.115 \times 5.22}{264.6} - \frac{1}{6} \right)$$

$$= 10.75$$

$$\frac{\text{Gap}}{\text{Mean span}} = \frac{4.44}{28.75} = .1545$$

$$\frac{\text{Lower span}}{\text{Upper span}} = \frac{26.0}{31.5} = .825$$

$$s = .53 \quad (\text{Fig. 4})$$

Authority: Page 78, Instructions for Airplane Designers,

Note: Area under fuselage assumed to be that of XP3A, given in Air Corps Information Circular No. 629, viz., 12.6 sq.ft.

DETERMINATION OF CORRECTED  $\frac{I}{D}$  FOR CELLULE:

$$\text{Corrected } \frac{I}{D} \text{ cellule} = \frac{\frac{I}{D} \text{ model} \times B \times K_v}{I}$$

$$B = \frac{125 \left( \frac{C_L}{C_D} + \frac{C_D}{C_L} + 2 \right)}{A} - \frac{I}{\text{Air model}} \quad \left( \right)$$

A = (total wing area area under fuselage) - 264.6 sq.ft.

$$B = \frac{125 \left( \frac{4.115}{2.22} + \frac{2.22}{4.115} + 2 \right)}{264.6} - \frac{I}{6} = 10.75$$

$$\frac{\text{Gap}}{\text{Mean span}} = \frac{4.44}{28.75} = .1545$$

$$\frac{\text{Lower span}}{\text{Upper span}} = \frac{26.0}{31.5} = .825$$

a = .53 (Fig. )

Authority: Page 78, Instructions for Airplane Designers.

Note: Area under fuselage assumed to be that of XP3A, given in Air Corps Information Circular No. 629, viz., 12.6 sq.ft.

DETERMINATION OF CORRECTED  $\frac{L}{D}$  FOR CELLULE: (Cont'd)

Airfoil $\alpha$	Airfoil $K_y$	1	Corrected $\frac{L}{D}$ (cellule)
		$\frac{D}{L} \text{ airfoil} + B \times K_y$	
-6.0	-.0001536	$\frac{1}{-.180 + (10.75 \times -.0001536)}$	= -5.5
-4.5	.0001152	$\frac{1}{.238 + (10.75 \times .0001152)}$	= 4.18
-3.0	.0004275	$\frac{1}{.0713 + (10.75 \times .0004275)}$	= 13.18
-1.5	.0006860	$\frac{1}{.0518 + (10.75 \times .000686)}$	= 16.9
0	.0009830	$\frac{1}{.0448 + (10.75 \times .000983)}$	= 18.05
1.5	.0012820	$\frac{1}{.0455 + (10.75 \times .001282)}$	= 16.85
3.0	.001540	$\frac{1}{.0478 + (10.75 \times .00154)}$	= 15.53
6.0	.002098	$\frac{1}{.0567 + (10.75 \times .002098)}$	= 12.6
9.0	.00265	$\frac{1}{.0677 + (10.75 \times .00265)}$	= 10.4
12.0	.00315	$\frac{1}{.080 + (10.75 \times .00315)}$	= 8.78
15.0	.00350	$\frac{1}{.093 + (10.75 \times .00350)}$	= 7.65
18.0	.003283	$\frac{1}{.1644 + (10.75 \times .003283)}$	= 5.01
21.0	.00277	$\frac{1}{.2725 + (10.75 \times .00277)}$	= 3.31

DETERMINATION OF CORRECTED  $\frac{I}{D}$  FOR CELLULOSE: (Cont'd)

Corrected $\frac{I}{D}$ (celulose)	$\frac{D}{I}$ at 10.75 x $\frac{K}{V}$	$\frac{I}{D}$ at 10.75 x $\frac{K}{V}$	$\frac{I}{D}$ at 10.75 x $\frac{K}{V}$
2.31	.3425 (10.75 x .0024)	1.644 (10.75 x .00328)	.0024
2.01	.3425 (10.75 x .00328)	.093 (10.75 x .00320)	.00328
1.69	.3425 (10.75 x .00320)	.080 (10.75 x .00312)	.00320
1.38	.3425 (10.75 x .00312)	.0677 (10.75 x .00292)	.00312
1.04	.3425 (10.75 x .00292)	.0567 (10.75 x .00268)	.00292
1.25	.3425 (10.75 x .00268)	.0478 (10.75 x .0024)	.00268
1.68	.3425 (10.75 x .0024)	.0425 (10.75 x .00188)	.0024
18.02	.3425 (10.75 x .00188)	.0448 (10.75 x .00088)	.00188
16.9	.3425 (10.75 x .00088)	.0318 (10.75 x .00068)	.00088
13.18	.3425 (10.75 x .00068)	.0713 (10.75 x .000425)	.00068
4.18	.3425 (10.75 x .000425)	.038 (10.75 x .000125)	.000425
2.2	.3425 (10.75 x .000125)	-.180 (10.75 x -.000125)	.000125



DETERMINATION OF RELATIVE EFFICIENCY OF WINGS - e :

$$s = \frac{\text{chord, lower}}{\text{chord, upper}} = \frac{C_2}{C_1} = \frac{49.4}{62.6} = .79$$

$$\frac{G}{C_1} = \frac{\text{gap}}{\text{chord, upper}} = \frac{53.312}{62.6} = .853$$

$$r = \frac{\text{span, lower}}{\text{span, upper}} = \frac{26.0}{31.5} = .825$$

$$\text{Stagger} = 32.50$$

$$e_{(r = .825)} = .825 e_{(r = 1)} + \frac{(1 - .825)}{(2 - Ca')} \text{ (Fig. 6 )}$$

$$e_{(r = 1)} = 1.51 \quad \text{(Fig. 6 )}$$

$$Ca' = \frac{\text{Lift of upper wing in biplane}}{\text{Lift of upper wing as monoplane}} = 1.2025 \quad \text{(Fig. 7 )}$$

$$e_{(r = .825)} = (.825 \times 1.51 + \frac{(1 - .825)}{(2 - 1.2025)}) = 1.47$$

Authority: A.D.W. 900

DETERMINATION OF RELATIVE EFFICIENCY OF LINES

$$e = \frac{\text{chord, upper}}{\text{chord, lower}} = \frac{C}{c} = \frac{49.4}{82.6} = .59$$

$$\frac{C}{c} = \frac{\text{chord, upper}}{\text{asp}} = \frac{82.6}{53.313} = .823$$

$$r = \frac{\text{span, upper}}{\text{span, lower}} = \frac{31.3}{36.0} = .869$$

$$2.2832 = 36.50$$

$$e(r = .823) = .823 \quad (r = 1) \quad \frac{(1 - .823)}{.8 - .68} \quad (.314)$$

$$e(r = 1) = 1.31 \quad (.314)$$

$$C = \text{Lift of upper wing as monoplane} = 1.3032 \quad \text{Lift of upper wing in biplane} = 1.3032 \quad (.314)$$

$$e(r = .823) = (.823 \times 1.31) \frac{(1 - .823)}{.8 - 1.3032} = 1.44$$

Authority: A.D.N. 200

DETERMINATION OF EQUIVALENT MONOPLANE ASPECT RATIO OF CELLULE:

$A' = \text{area of upper wing} = 158 \text{ sq.ft.}$

$A'' = \text{area of lower wings} + \text{portion cut out by fuselage},$   
 $= 94 + 12.6 = 106.6 \text{ sq.ft.}$

$A''' = \text{area of lower wings} + 50\% \text{ of portion cut out by fuselage}$   
 $= 94 + 6.3 = 100.3 \text{ sq.ft.}$

$$a = \frac{e A'}{e A' + A''} = \frac{1.47 \times 158}{(1.47 \times 158) + 100.3} = .70$$

$$\frac{G}{b} = \frac{\text{Gap}}{\text{longer span}} = \frac{4.44}{31.5} = 1.41$$

$$K_{(r=.8)} = 1.054$$

$$K_{(r=.9)} = 1.08$$

$$r = \frac{\text{span, lower}}{\text{span, upper}} = \frac{26.0}{31.5} = .825$$

$$K_{(r = .825)} = 1.06 = \frac{\text{Span of equivalent monoplane}}{\text{upper span}}$$

$$\text{Span of equivalent monoplane} = (31.5 \times 1.06) = 33.4$$

$$\text{Equivalent monoplane A.R.} = \frac{(33.4)^2}{(12.6 + 252)} = 4.22$$

References: Air Corps I.C. #607 and #629.

DETERMINATION OF EQUIVALENT MONOPLANE ASPECT RATIO OF CONTROL:

"A" = area of lower wings - 50% of portion cut out by fuselage  
 = 94 12.6 = 108.6 sq.ft.  
 "A'" = area of upper wing = 128 sq.ft.  
 "A" = area of lower wings - portion cut out by fuselage.

$$\frac{S = 0.4'}{S' = A'} = \frac{1.47 \times 128}{(1.47 \times 128) - 100.3} = .70$$

$$\frac{C}{S} = \frac{\text{lower span}}{\text{gap}} = \frac{31.5}{4.44} = 1.41$$

$$K(x=.8) = 1.024$$

$$K(x=.9) = 1.08$$

$$x = \frac{\text{span, upper}}{\text{span, lower}} = \frac{31.5}{36.0} = .875$$

$$K(x) = (.825) \quad 1.06 \quad \frac{\text{span of equivalent monoplane}}{\text{upper span}}$$

$$\text{span of equivalent monoplane} = (31.5 \times 1.06) = 33.4$$

$$\text{Equivalent monoplane A.R.} = \frac{(33.4)}{(12.6)} = 4.25$$

References: Air Corps I.C. 1607 and 1622.

DETERMINATION OF CELLULE ANGLES OF ATTACK:

Equivalent monoplane aspect ratio 4.22

$$\alpha_{AR}^{\circ} = \alpha_{AR \text{ model}}^{\circ} - (125 \times 57.3 K_y) \left( \frac{1}{AR \text{ model}} - \frac{1}{AR} \right)$$

$$\begin{aligned} \alpha_{4.22}^{\circ} &= -6.0 - (125 \times 57.3 \times -.0001536) \left( \frac{1}{6} - \frac{1}{4.22} \right) = -6.08 \\ &= -4.5 + (501 \times .0001152) = -4.44 \\ &= -3.0 + (501 \times .0004275) = -2.78 \\ &= -1.5 + (501 \times .000686) = -1.15 \\ &= 0 + (501 \times .000983) = 0.5 \\ &= 1.5 + (501 \times .001283) = 2.15 \\ &= 3.0 + (501 \times .00154) = 3.77 \\ &= 6.0 + (501 \times .0020) = 7.06 \\ &= 9.0 + (501 \times .00265) = 10.33 \\ &= 12.0 + (501 \times .00315) = 13.59 \\ &= 15.0 + (501 \times .0035) = 16.77 \\ &= 18.0 + (501 \times .00328) = 19.65 \\ &= 21.0 + (501 \times .00277) = 22.39 \end{aligned}$$

Authority : A.D.M. 1061, page 17.

# DETERMINATION OF CERTAIN ANGLES OF ATTACK:

Equivalent monoplane aspect ratio 4.25

$$AR = AR_{model} - (1.25 \times 27.3 K_v) \left( \frac{1}{AR_{model}} - \frac{1}{AR} \right)$$

$$4.25 = -6.0 - (1.25 \times 27.3 \times -.001336) \left( \frac{1}{6} - \frac{1}{4.25} \right) = -6.08$$

=	-4.4	(.001132 x 201)	=	-4.44
=	-3.0	(.0004272 x 201)	=	-3.78
=	-1.5	(.000686 x 201)	=	-1.12
=	0	(.000983 x 201)	=	0.2
=	1.5	(.001283 x 201)	=	3.12
=	3.0	(.001584 x 201)	=	3.77
=	6.0	(.0020 x 201)	=	7.06
=	9.0	(.00265 x 201)	=	10.23
=	12.0	(.00312 x 201)	=	13.29
=	15.0	(.0035 x 201)	=	16.77
=	18.0	(.00388 x 201)	=	19.62
=	21.0	(.00427 x 201)	=	22.39

DETERMINATION OF C.P. :

$$\text{C.P. \% M.A.C.} = .25 - \frac{K_m \text{ 25\%}}{K_n}; \text{ where } K_n = K_y \cos \alpha + K_x \sin \alpha$$

Cellule $\alpha$	$K_y$	Cellule $K_x$	$K_m$ about 25% of chord,	$K_n$	C.P. % M.A.C.
-6.08	-.0001536	.0000279	-.0002125	-.000156	-111.0
-4.44	.0001152	.0000275	-.000205	.000113	206.
-2.78	.0004275	.0000324	-.0001997	.0004245	72
-1.15	.000686	.0000406	-.0001997	.000685	54.1
0.5	.000983	.0000544	-.0001792	.000983	43.2
2.15	.001282	.0000760	-.000151	.001284	36.75
3.77	.001540	.0000992	-.000187	.001548	37.1
7.06	.002098	.0001665	-.0002022	.002106	34.6
10.33	.00265	.000255	-.0001383	.002658	30.2
13.59	.00315	.000359	-.0000922	.00315	27.9
16.77	.00350	.0004575	-.000197	.003484	30.65
19.65	.003283	.000656	-.0001434	.00332	29.3
22.39	.00277	.000837	-.000123	.00288	29.3

By extrapolation of airplane characteristic curves:

-7.0	-.0032	.000029	-.0002125	-.0003215	-41.0
-8.0	-.00049	.000030	-.000213	-.000489	-18.5
-9.0	-.00066	.000031	-.000214	-.000657	- 7.5
-10.0	-.00083	.000037	-.000215	-.000825	- 1.1



# DETERMINATION OF C.P. :

C.P. & M.A.C. SS -  $K_m$  SS<sub>g</sub> ; where  $K_m$   $K_y$  cos.  $K$  sin

Celulose	$K_y$	Celulose $K_x$	SS <sub>g</sub> of $K_m$ about	$K_m$	C.P. & M.A.C.
22.32	.00277	.000837	-.000123	.00288	22.3
19.62	.002282	.000626	-.000124	.00232	22.3
16.77	.00220	.0004272	-.000127	.002484	20.62
13.22	.00212	.000322	-.0000222	.00212	27.2
10.32	.00262	.000222	-.0001282	.002628	20.2
7.06	.002028	.0001662	-.0002022	.002106	24.6
3.77	.001240	.0000222	-.000187	.001248	27.1
2.12	.001282	.0000760	-.000121	.001284	26.72
0.2	.000282	.0000244	-.000122	.000282	42.2
-1.12	.000686	.0000406	-.0001227	.000682	24.1
-2.78	.0004272	.0000224	-.0001227	.0004242	72
-4.44	.000122	.0000272	-.000202	.00012	206
-6.08	-.0001226	.0000272	-.0002122	-.000126	-111.0

By extrapolation of airplane characteristic curves:

-10.0	-.00082	.000027	-.000212	-.000822	-1.1
-2.0	-.00066	.000021	-.000214	-.000627	-7.2
-8.0	-.00042	.000020	-.000212	-.000422	-12.2
-7.0	-.0022	.000022	-.0002122	-.000212	-41.0

REDETERMINATION OF C.P. FROM SMOOTH  $K_M$  L.E.:

$$K_r = \sqrt{K_y^2 + K_x^2}$$

Cellule $\alpha$	Inches from L.E. to C.P.	$K_r$	$K_M$ L.E.	$K_M$ L.E. from smooth curve.	C.P. %M.A.C. Redetermined.
-6.08	-65.15	-.000156	-.000173	-.000173	-111.0
-4.44	121.0	.000118	-.000243	-.000243	206
-2.78	42.25	.000429	-.000309	-.000309	72
-1.15	31.75	.000687	-.0003715	-.000 75	54.6
0.5	25.35	.000985	-.000426	-.000445	45.2
2.15	21.55	.001283	-.0004715	-.000505	39.3
3.77	21.8	.001543	-.0005725	-.00057	36.9
7.06	20.3	.002105	-.000728	-.00069	32.8
10.33	17.72	.00266	-.000803	-.000803	30.2
13.59	16.4	.00317	-.000885	-.000916	28.9
16.77	18.0	.00353	-.001082	-.001082	30.65
19.65	17.2	.00335	-.000981	-.000981	29.3
22.39	17.15	.00278	-.000815	-.000815	29.3

By extrapolation of airplane characteristic curves:

-7.0	-24.05	-.000326	-.0001337	-.0001337	-41.0
-8.0	-10.86	-.000491	-.00009075	-.00009075	-18.5
-9.0	- 4.4	-.00066	-.0000495	-.0000495	- 7.5
10.0	.65	-.000831	-.000091	-.000091	- 1.1

Note: Redetermination of C.P. was made for the purpose of smoothing out basic moment data on Clerk Y airfoil, in order better to define curve of  $K_M$  c.g.

# REDETERMINATION OF C.P. FROM SMOOTH K L.E.

K<sub>1</sub> K<sub>2</sub> K<sub>3</sub>

Cellulose inches from L.E. to C.P. K<sub>1</sub>  
 from smooth K<sub>2</sub> L.E. K<sub>3</sub>  
 C.P. & M.A.C. Redetermined.

Curves					
-6.08	-0.00173	-0.000173	-0.000156	-65.15	-6.08
-4.44	-0.00243	-0.000243	-0.000118	121.0	-4.44
-3.78	-0.00309	-0.000309	-0.000429	42.25	-3.78
-1.15	-0.0075	-0.0003715	-0.000687	31.75	-1.15
0.5	-0.00445	-0.000426	-0.000385	25.35	0.5
2.15	-0.00205	-0.0004715	-0.001283	21.55	2.15
3.77	-0.0027	-0.000525	-0.001243	21.8	3.77
7.06	-0.00069	-0.000728	-0.002105	20.3	7.06
10.33	-0.000803	-0.000803	-0.00266	17.75	10.33
12.59	-0.000916	-0.000885	-0.00317	16.4	12.59
16.77	-0.001082	-0.001082	-0.00353	18.0	16.77
19.65	-0.000981	-0.000981	-0.00335	17.2	19.65
22.39	-0.000815	-0.000815	-0.00278	17.15	22.39

By extrapolation of airplane characteristic curves:

-7.0	-24.05	-0.000326	-0.0001337	-0.0001337	-41.0
-8.0	-10.86	-0.000491	-0.00009075	-0.00009075	-18.5
-9.0	-4.4	-0.00066	-0.0000495	-0.0000495	-7.5
10.0	.65	-0.000821	-0.000021	-0.0000021	-1.1

Note: Redetermination of C.P. was made for the purpose of smoothing out basic moment data on Clark Y airfoil, in order better to define curve of K c.g.

DETERMINATION OF  $\frac{K}{M}$  c.g.:

Cellule $\alpha$	$(\beta^\circ - \alpha^\circ)$	Inches from L.C. to C.P.	Inches from C.G. to vector	$K_M$ c.g.
-6.08	4.25°	-65.2	83.6	-.000222
-4.44	18.0°	121.0	89.4	-.00018
-2.78	7.25°	42.2	19.4	-.000142
-1.15	4.50°	32.0	10.2	-.0001193
.5	2.50°	26.5	5.5	-.0000923
2.15	1.50°	23.05	2.5	-.0000547
3.77	.50°	21.65	1.4	-.0000368
7.06	2.25°	19.25	.0	0
10.33	4.50°	17.72	-0.5	.0000227
13.59	7.25°	16.97	-0.25	.0000135
16.77	9.50°	18.0	1.5	-.0000902
19.65	8.25°	17.2	.2	.0000114
22.39	5.50°	17.2	-0.8	.0000379

By extrapolation of airfoil moment curve:

-7.0	2.0°	-24.05	44.75	-.0002485
-8.0	4.5°	-10.86	32.4	-.000271
-9.0	6.25°	-4.4	26.6	-.000299
10.0	7.25°	-0.65	23.1	-.000327

$(\beta^\circ - \alpha^\circ)$  = Angle between normal to chord and resultant vector,  $K_r$ .

# DETERMINATION OF $\frac{K}{M}$ C.B.

Celcius	( )	Inches from L.C. to C.P.	Inches from C.C. to vector	$\frac{K}{M}$ C.B.
22.39	2.20	17.2	-0.8	.0000379
19.65	8.25	17.2	.2	.0000114
16.77	9.20	18.0	1.2	-.0000902
13.29	7.25	16.97	-0.25	.0000132
10.32	4.20	17.72	-0.2	.0000227
7.06	2.25	19.22	0.	0
3.77	1.20	21.62	1.4	-.0000368
2.12	1.20	23.02	2.2	-.0000247
.2	2.20	26.2	2.2	-.0000223
-1.12	4.20	32.0	10.2	-.0001193
-2.78	7.25	42.2	19.4	-.000142
-4.44	18.0	121.0	29.4	-.00018
-6.08	4.25	-62.2	83.6	-.000222

By extrapolation of airfoil moment curve:

10.0	7.25	- 0.62	23.1	-.000227
9.0	6.25	- 4.4	26.6	-.000292
8.0	4.2	-10.86	32.4	-.000271
-7.0	2.0	-24.02	44.72	-.0002482

( ) - Angle between normal to chord and resultant vector,  $K_r$ .

DETERMINATION OF AIRPLANE  $K_y$  :

$$\text{Airplane } K_y = \text{Cellule } K_y + \frac{\text{Tail Load}}{A_w V^2}$$

$$\frac{\text{Tail Load}}{A_w V^2} = K_M \text{ c.g.} \times \frac{A_w V^2}{A_w V^2} \times \frac{C}{d} = K_M \text{ c.g.} \times \frac{58.7}{167.5} = .3504 K_M \text{ c.g.}$$

$$\alpha \quad \text{Cellule } ( \text{Cellule } K_y + .3504 K_M \text{ c.g.} ) = \text{Airplane } K_y$$

-6.08	( -.0001536	-.0000781 )	=	-.000232
-4.44	( .0001152	-.000064 )	=	.0000512
-2.78	( .0004275	-.0000506 )	=	.000377
-1.15	( .000686	-.0000401 )	=	.000646
0.5	( .000983	-.0000300 )	=	.000953
2.15	( .001282	-.0000199 )	=	.001262
3.77	( .00154	-.000012 )	=	.001528
7.06	( .002098	-.0000038 )	=	.002098
10.33	( .00265	0 )	=	.00265
13.59	( .00315	0 )	=	.00315
16.77	( .00350	-.0000105 )	=	.00349
19.65	( .003283	0 )	=	.003283
22.39	( .00277	.000015 )	=	.00278

By extrapolation of Airplane  $K_y$  curve:

-7.0	-.000409
-8.0	-.000587
-9.0	-.000767
-10.0	-.000948

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-10.0  
-9.0  
-8.0  
-7.0  
-0.00409  
-0.00387  
-0.00367  
-0.00348

By extrapolation of Airplane  $K_y$  curve:

22.32	(.00277	.000015	(.00278
19.65	(.00283	0	(.00283
16.77	(.00320	-0.000102	(.00320
13.52	(.00315	0	(.00315
10.33	(.00262	0	(.00262
7.06	(.00208	-0.000028	(.00208
3.77	(.00154	-0.000012	(.00154
2.12	(.001282	-0.000012	(.001282
0.5	(.000983	-0.000020	(.000983
-1.12	(.000686	-0.0000401	(.000686
-2.78	(.0004272	-0.000206	(.000427
-4.44	(.000125	-0.00064	(.000125
-6.08	(-.0001536	-0.000781	(-.000153

Cellule  
(Cellule K 3204 K c.e.)  
Airplane  $K_y$

$\frac{A_w}{V}$  Tail Load

K c.e.

$\frac{A_w}{V} \times \frac{A_w}{V}$

$\frac{C}{B}$

K c.e.

$\frac{58.7}{16.5} \times$

.3204 K c.e.

DETERMINATION OF AIRPLANE  $K_y$



DETERMINATION OF EQUIVALENT FLAT PLATE AREA:

$$V_{\max} = 157 \text{ m.p.h.}$$

$$N = 1950 \text{ r.p.m.}$$

$$D = 8.665 \text{ ft.}$$

$$\frac{V}{N D} = \frac{157 \times 1.47}{\frac{1950}{60} \times 8.667} = .82$$

$$\eta \text{ at max efficiency} = .84$$

$$\text{B.H.P.} = 435$$

$$\text{H.P.}_a = \text{B.H.P.} \times \eta = \text{H.P.}_r$$

$$\text{H.P.}_r = 435 \times .84 = 365.$$

$$\text{Total Drag} = \frac{375 \times \text{H.P.}_r}{V} = \frac{375 \times 365}{157} = 872 \text{ lbs.}$$

$$W = K_y A V^2$$

$$2796 = K_y \times 252 \times (157)^2$$

$$\text{Airplane } K_y = .00045 = \text{Cellule } K_y + \frac{58.7}{167.5} \times K_M \text{ c.g.}$$

$$\text{For Airplane } K_y = .00045, \alpha = -2.3 \text{ (from airplane } K_y \text{ curve).}$$

$$\text{At } \alpha = -2.3^\circ, K_M \text{ c.g.} = -.0001375$$

$$\text{Cellule } K_y = .00045 - (.3504 \times -.0001375) = .000498$$

$$\text{Cellule } K_x = .000033 \quad (\text{from cellule polar}).$$

$$\text{Wing cellule drag} = .000033 \times 252 \times (157)^2 = 205 \text{ lbs.}$$

$$\text{Parasite drag} = (872 - 205) = 667 \text{ lbs.}$$

$$A_e \times 32.7 \times \left(\frac{157}{100}\right)^2 = 667$$

$$A_e = 8.28 \text{ sq. ft.}$$

DETERMINATION OF EQUIVALENT FLAT PLATE AREA:

$V_{max} = 157 \text{ m.p.h.}$   
 $N = 1950 \text{ r.p.m.}$   
 $D = 8.665 \text{ ft.}$   
 $N/D = \frac{1950}{8.665} \times \frac{1.47}{157} = 82$   
 $\therefore \text{at max efficiency } 84$   
 $B.H.P. = 435$   
 $H.P. = B.H.P. \times 0.84 = 365$   
 $H.P. = 435 \times 0.84 = 365$   
 $\text{Total Drag} = \frac{375 \times H.P.}{V} = \frac{375 \times 365}{157} = 875 \text{ lbs.}$   
 $W = K_v A V^3$   
 $2796 = K_v \times 252 \times (157)$   
 $Airplane K_v = \frac{2796}{0.0045 \times 252 \times 157} = 167.5$   
 $\therefore K_v = 167.5$   
 $\text{For Airplane } K_v = 0.0045 \times -2.3 = -0.001375$   
 $\text{At } -2.3 \text{ K v.e.} = -0.001375$   
 $\text{Cellule } K_v = 0.0045 - (0.3504 \times -0.001375) = 0.00498$   
 $\text{Cellule } K_x = 0.00033$   
 $\text{(from cellule polar)}$   
 $\text{Wing cellule drag} = 0.00033 \times 252 \times (157) = 205 \text{ lbs.}$   
 $\text{Parasite drag} = (875 - 205) = 670 \text{ lbs.}$   
 $A = \frac{100}{670} \times 32.7 \times (157) = 667$   
 $A = 8.28 \text{ sq. ft.}$

DETERMINATION OF AIRPLANE  $K_X$  :

$$\text{Airplane } K_X = K_X (\text{cellule}) + \frac{A_e}{A_w} \times .00327 = K_X (\text{cellule}) +$$

$$\frac{8.28}{252} \times .00327 = K_X (\text{cellule}) + .0001075$$

Cellule $\alpha$	$K_X$ (cellule) + .0001075	= $K_X$ (airplane)
-6.08	.0000279 + .0001075	= .0001354
-4.44	.0000275 + .0001075	= .0001350
-2.78	.0000324 + .0001075	= .0001399
-1.15	.0000406 + .0001075	= .0001481
0.5	.0000544 + .0001075	= .0001619
2.15	.000076 + .0001075	= .0001835
3.77	.0000992 + .0001075	= .0002067
7.06	.0001665 + .0001075	= .0002740
10.36	.000255 + .0001075	= .0003625
13.59	.000359 + .0001075	= .0004665
16.77	.0004575 + .0001075	= .000565
19.65	.000656 + .0001075	= .0007635
22.39	.000837 + .0001075	= .0009445

By extrapolation of Airplane  $K_X$  curve:

-7.0	.000136
-8.0	.000137
-9.0	.000138
-10.0	.000144

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-10.0	.000144
-9.0	.000138
-8.0	.000137
-7.0	.000136

By extrapolation of Airplane K<sub>x</sub> curve:

Celcius	K <sub>x</sub> (celcius)	.0001075	K <sub>x</sub> (airplane)
22.39	.000837	.0001075	.0009445
19.65	.000656	.0001075	.0007635
16.77	.0004575	.0001075	.000565
13.59	.000359	.0001075	.0004665
10.36	.000255	.0001075	.0003655
7.06	.0001665	.0001075	.0002740
3.77	.0000995	.0001075	.0002067
2.15	.000076	.0001075	.0001835
0.5	.0000344	.0001075	.0001619
91.15	.0000406	.0001075	.0001481
-2.78	.0000324	.0001075	.0001399
-4.44	.0000275	.0001075	.0001350
-6.08	.0000279	.0001075	.0001354

$$\frac{8.58}{22} \times .00327 \text{ K}_x (\text{celcius}) = .0001075$$

$$\frac{\text{Airplane K}_x - \text{K}_x (\text{celcius})}{\text{Aw}} \times .00327 \text{ K}_x (\text{celcius})$$

DETERMINATION OF AIRPLANE K<sub>x</sub> :

DETERMINATION OF VELOCITIES:

$$V = \sqrt{\frac{W}{A}} \sqrt{\frac{\cos \beta}{K_y}} = \sqrt{\frac{2796}{252}} \sqrt{\frac{\cos \beta}{K_y}} = 3.33 \sqrt{\frac{\cos \beta}{K_y}}$$

$$= 3.33 \sqrt{\frac{\sin \beta}{K_x}}$$

Cellule Airplane Flight path

$\alpha$	$\frac{L}{D}$	angle $\beta = \cot^{-1} \frac{L}{D}$	$\frac{L}{D}$	$\cos \beta$	Airplane $K_y$	$\frac{\cos \beta}{K_y}$	V
-6.08	-1.713	(-)30-16		.8637	-.000232	3725	203.3
-4.7	0	90	sin	1.000	Kx.000135	7400	286.5
-4.44	.379	69-14		.3546	.0000512	6920	277
-2.78	2.69	20-23		.9374	.000377	2485	166
-1.15	4.36	12-55		.9747	.000646	1510	129.5
.5	5.88	9-39		.9858	.000953	1034	107.2
2.15	6.87	8-17		.9896	.001262	784	93.2
3.77	7.34	7-45		.9909	.001528	648	84.1
7.06	7.65	7-27		.9916	.002098	472	72.3
10.33	7.31	7-47		.9908	.00265	374	64.5
13.59	6.75	8-25		.9892	.00315	314	59.4
16.77	6.13	9-16		.9869	.00349	283	56.1
19.65	4.30	13-05		.9740	.003283	297	57.4
22.39	2.945	18-45		.9469	.00278	341	61.5

By extrapolation of Airplane characteristic curves:

-7.0	-3.00	(-)18-26		.9487	-.000409	2320	160.5
-8.0	-4.265	(-)13-12		.9736	-.000587	1660	135.8
-9.0	-5.54	(-)10-14		.9841	-.000767	1283	120
-10.0	-6.56	(-) 8-40		.9886	-.000948	1043	107.5

# DETERMINATION OF VELOCITIES:

$V$	$\frac{W}{V}$	$\frac{\cos \gamma}{K}$	$\frac{\sin \delta}{K}$	$\frac{\cos \delta}{K}$	$\frac{\sin \gamma}{K}$
22.39	5.945	18-45	.9469	.00278	341
19.65	4.30	18-05	.9740	.00283	327
16.77	6.13	9-16	.9869	.00349	283
13.59	6.75	8-25	.9892	.00315	314
10.33	7.31	7-47	.9908	.00265	374
7.06	7.65	7-27	.9916	.00298	475
3.77	7.34	7-45	.9909	.001528	648
2.15	6.87	8-17	.9896	.001565	784
5	5.88	9-39	.9858	.000923	1034
-1.15	4.36	12-25	.9747	.000646	1510
-2.78	5.69	20-23	.9374	.000377	2485
-4.44	3.79	62-14	.3546	.0000515	6250
-4.7	0	90	sin 1.000	Kx.000135	7400
-6.08	-1.713	(-) 30-16	.8637	-.000325	3725

By extrapolation of Airplane characteristic curves;

-10.0	-6.56	(-) 8-40	.9886	-.000948	1043
-9.0	-5.54	(-) 10-14	.9841	-.000767	1283
-8.0	-4.365	(-) 13-15	.9736	-.000587	1660
-7.0	-3.00	(-) 18-26	.9487	-.000409	2320

DETERMINATION OF TAIL LOADS:

$$\text{Normal Tail Load} = K_M \text{ c.g.} \times A_w \times \frac{C}{\bar{a}} \times V^2 = K_M \text{ c.g.} \times \frac{252 \times 58.7}{167.5} \times V^2$$

$$= 88.3 \times V^2 \times K_M \text{ c.g.}$$

$$\text{Dynamic Tail Load} = \text{Normal Tail Load} \times \left( \frac{V_{\max}}{V} \right)^2$$

Cellule	$K_M \text{ c.g.}$	$V^2$	$\left( \frac{V_{\max}}{V} \right)^2$	Dynamic Tail Load.	Normal Tail Load.
-6.08	-.000223	41350	1.987	1620	-815
-4.7	-.0001875	82000	1.0	1359	-1359
-4.44	-.0001825	76700	1.07	1323	-1236
-2.78	-.0001445	27550	2.97	1047	-352
-1.15	-.0001147	16800	4.89	831	-170
.5	-.0000855	11500	7.14	620	-86.8
2.15	-.0000567	8680	9.45	410	-43.4
3.77	-.0000342	7070	11.60	248	-21.4
7.06	-.00001075	5225	15.7	88	-5.6
10.33	0	3165	19.95	0	0
13.59	0	3530	23.25	0	0
16.77	-.0000301	3140	26.1	218	-8.3
19.65	0	3300	24.9	0	0
22.39	.0000429	3780	21.7	311	14.3

By extrapolation of Airplane characteristic curves;

-7.0	-.000253	25770	3.185	1775	-575
-8.0	-.000277	18450	4.45	2005	-451
-9.0	-.0003055	14400	5.7	2220	-389
-10.0	-.000336	11560	7.1	2440	-343

Authority: A.D.M. 1061

Authority: A.D.M. 1061

-10.0	-0.00336	11500	7.1	2440	-343
-9.0	-0.003055	14400	5.7	2250	-389
-8.0	-0.00277	18450	4.45	2005	-431
-7.0	-0.00253	22770	3.185	1775	-375

By extrapolation of airplane characteristic curves:

22.39	0.000452	3780	21.7	311	14.3
19.65	0	3300	24.2	0	0
16.77	-0.000301	3140	26.1	218	-8.3
13.59	0	3230	23.25	0	0
10.33	0	3165	19.95	0	0
7.06	-0.0001075	5235	15.7	88	-5.6
3.77	-0.000345	7070	11.60	248	-21.4
2.15	-0.000567	8680	9.45	410	-43.4
5	-0.000855	11500	7.14	650	-86.8
-1.15	-0.001147	16800	4.89	831	-170
-3.78	-0.001445	27550	3.97	1047	-335
-4.44	-0.001855	46700	1.07	1323	-1336
-4.7	-0.001875	82000	1.0	1359	-1359
-6.08	-0.00253	41350	1.987	1650	-815

$\frac{V}{V_{max}}$        $V$        $K \text{ c.g.}$   
 Dynamic Tail Load      Normal Tail Load      Normal Tail Load

Coefficient

$\left( \frac{V}{V_{max}} \right) \times \text{Normal Tail Load}$

88.3 x V x K c.g.

Normal Tail Load K c.g. x  $\frac{C}{5}$  x  $V$  x K c.g. x 2525 x 28.7 x V  
 167.5

DETERMINATION OF TAIL LOADS:



CURTISS F6C-4 FIGHTING AIRPLANE  
TAIL LOAD COMPUTATIONS.

Tail load computations were repeated herein, assuming a gross weight, and C.G. location of the airplane, pressure distribution test of which, are recorded in N.A.C.A. Report No. 307. Airfoil data forming the basis for these computations have been corrected for Wall Interference.

TAIL LOAD COMPUTATIONS.  
CURTIS REC-4 FIGHTING AIRPLANE

been corrected for wall interference.  
Aircraft data forming the basis for these computations have  
tribution test of which, are recorded in N.A.C.A. Report No. 307.  
gross weight, and C.G. location of the airplane, pressure dis-  
Tail load computations were repeated herein, assuming a

BASIC DATA ON F6C-4 AIRPLANE:

Gross Weight (W) -	2580
Wing Section -	Clark "Y"
M.A.C. (C) -	58.7"
Span, upper -	31.5'
Span, lower -	26.0'
Chord, upper, (average) -	62.6"
Chord, lower, (average) -	49.4"
Gap -	53.31"
Stagger, at L.E., at fuselage -	38.5"
C.G. % M.A.C. -	33.4%
C.G. %M.A.C. below -	28.1%
Area horizontal tail surfaces -	32.9 sq.ft.
Total Wing Area -	252 sq.ft.
Distance from C.G. to tail post - (d)	167.9"
Area, upper wing -	158 sq.ft.
Area, lower wing -	94 sq.ft.
Maximum speed, sea level -	157 m.p.h.
B.H.P. 435 at 1950 r.p.m.	
Diameter of propeller	8.665 ft.
L.E.M.A.C. 23.6" ahead L.E.L.W.	

Note: Gross weight and C.G. location given above, is as specified for F6C-4 Pressure Distribution Tests, recorded in N.A.C.A. Report No. 307.

BASIC DATA ON P6C-4 AIRPLANE:

2580	Gross Weight (W) -
Clark "Y"	Wing section -
58.7"	M.A.C. (C) -
31.5'	Span, upper -
26.0'	Span, lower -
62.6"	Chord, upper, (average) -
49.4"	Chord, lower, (average) -
53.31"	Gap -
38.5"	Stagger, at I.E., at fuselage -
33.4"	C.G. & M.A.C. -
38.1"	C.G. & M.A.C. below -
32.9 sq.ft.	Area horizontal tail surfaces -
252 sq.ft.	Total Wing Area -
167.9"	Distance from C.G. to tail post - (4)
158 sq.ft.	Area, upper wing -
94 sq.ft.	Area, lower wing -
157 m.p.h.	Maximum speed, sea level -
	B.H.P. 435 at 1950 r.p.m.
8.665 ft.	Diameter of propeller
	I.E.M.A.C. 28.6" ahead I.E.I.W.

Note: Gross weight and C.G. location given above, is as specified for P6C-4 Pressure Distribution Tests, recorded in N.A.C.A. Report No. 307.

CLARK "Y" CHARACTERISTICS:

$\alpha$	$C_L$	$C_D$	$C_D$	$C_{M1.E.}$	C.P.	$\frac{L}{D}$	$\frac{D}{L}$
-6.02	-0.060	.0108	.0106	-.068	-1.117	-5.55	-.180
-4.48	.045	.0107	.0106	-.091	1.99	4.21	.238
-2.94	.167	.0121	.0106	-.120	.720	13.8	.0724
-1.40	.268	.0144	.0106	-.145	.541	18.6	.0537
.15	.384	.0182	.0103	-.166	.432	21.1	.0474
1.69	.501	.0245	.0111	-.185	.368	20.4	.0489
3.23	.602	.0312	.0119	-.224	.371	19.3	.0518
6.31	.819	.0508	.0152	-.284	.346	16.1	.062
9.39	1.034	.0770	.0201	-.312	.302	13.4	.0745
12.47	1.231	.1085	.0280	-.360	.294	11.4	.0881
15.52	1.367	.1395	.0403	-.415	.306	9.8	.102
18.49	1.283	.2217	.1342	-.378	.294	5.8	.1725
21.41	1.081	.3025	.2402	-.328	.293	3.58	.2795

Authority: N.A.C.A. Report No. 331 (High Density Tunnel)  
data corrected for Wall Interference).

CLARK "Y" CHARACTERISTICS:

D	I	C.P.	C.M.F.	CD	CD	CI	
180	-5.55	-1.117	-0.068	.0106	.0108	-0.060	-6.05
.528	4.21	1.92	-0.091	.0106	.0107	.045	-4.48
.0724	13.8	.720	-0.120	.0106	.0121	.167	-3.24
.0237	18.6	.341	-0.145	.0106	.0144	.268	-1.40
.0474	21.1	.432	-0.166	.0103	.0182	.384	.12
.0489	20.4	.368	-0.182	.0111	.0242	.501	1.69
.0218	19.3	.271	-0.224	.0112	.0312	.602	3.23
.062	16.1	.346	-0.284	.0122	.0508	.812	6.21
.0745	13.4	.302	-0.312	.0201	.0770	1.034	9.32
.0881	11.4	.294	-0.360	.0280	.1082	1.221	12.47
.102	9.8	.306	-0.412	.0402	.1222	1.367	15.22
.1722	5.8	.294	-0.378	.1242	.2217	1.282	18.42
.2722	3.22	.222	-0.328	.2402	.3022	1.081	21.41

Authority: W.A.C.A. Report No. 221 (High Density Tunnel)  
 data corrected for wall interference).

DETERMINATION OF CORRECTED  $\frac{L}{D}$  FOR CELLULE :

Airfoil

$\alpha$	$K_y$	$\frac{1}{D_{airfoil} + B \times K_y}$	= corrected $\frac{L}{D}$ (cellule)
-6.02	-.0001536	$\frac{1}{-.180 + (10.75 \times -.0001536)}$	= -5.5
-4.48	.0001152	$\frac{1}{.238 + (10.75 \times .0001152)}$	= 4.18
-2.94	.0004275	$\frac{1}{.0724 + (10.75 \times .0004275)}$	= 13.0
-1.40	.000686	$\frac{1}{.0537 + (10.75 \times .000686)}$	= 16.38
.15	.000983	$\frac{1}{.0474 + (10.75 \times .000983)}$	= 17.24
1.69	.001282	$\frac{1}{.498 + (10.75 \times .001282)}$	= 15.95
3.23	.001540	$\frac{1}{.518 + (10.75 \times .001540)}$	= 14.63
6.31	.002098	$\frac{1}{.062 + (10.75 \times .002098)}$	= 11.84
9.39	.00265	$\frac{1}{.0745 + (10.75 \times .002650)}$	= 9.7
12.47	.003150	$\frac{1}{.0881 + (10.75 \times .003150)}$	= 8.2
15.52	.00350	$\frac{1}{.102 + (10.75 \times .00350)}$	= 7.17
18.49	.003283	$\frac{1}{.1725 + (10.75 \times .003283)}$	= 4.81
21.41	.002770	$\frac{1}{.2795 + (10.75 \times .00277)}$	= 3.23

Note: Value of the factor "B" same as for previous calculation for F6C-4 Airplane.

DETERMINATION OF CORRECTED I  
FOR CELLULOSE :

Corrected (celcius)	Barrel	Barrel	Barrel
3.23	1.142 x (10.75 x 0.00383)	0.00383	31.41
4.81	1.108 x (10.75 x 0.00350)	0.00350	18.48
7.17	1.081 x (10.75 x 0.003150)	0.003150	12.52
8.5	1.045 x (10.75 x 0.00250)	0.00250	12.47
9.7	1.008 x (10.75 x 0.00208)	0.00208	9.38
11.84	0.918 x (10.75 x 0.001940)	0.001940	6.31
14.68	0.798 x (10.75 x 0.001583)	0.001583	3.33
15.95	0.744 x (10.75 x 0.000983)	0.000983	1.68
17.24	0.627 x (10.75 x 0.00068)	0.00068	1.15
18.38	0.524 x (10.75 x 0.000425)	0.000425	0.41
19.0	0.338 x (10.75 x 0.000112)	0.000112	0.24
21.4	0.180 x (10.75 x 0.0001236)	0.0001236	0.48
22.5			0.05

Note: Value of the factor "B" same as for previous calculation for HEC-4 Alignment.



DETERMINATION OF CELLULOSE ANGLES OF ATTACK:

Equivalent monoplane aspect ratio 4.22

$$\alpha^{\circ}_{AR} = \alpha^{\circ}_{AR \text{ model}} - (125 \times 57.3 K_y) \left( \frac{1}{AR \text{ model}} - \frac{1}{AR} \right)$$

$$\alpha^{\circ}_{4.22} = -6.02 - (125 \times 57.3 \times -.0001536) \left( \frac{1}{6} - \frac{1}{4.22} \right) = -6.1$$

-4.48	+	(501 x .0001152)	=	-4.42
-2.94	+	(501 x .0004275)	=	-2.72
-1.4	+	(501 x .000686)	=	-1.05
.15	+	(501 x .000983)	=	.65
1.69	+	(501 x .001282)	=	2.34
3.23	+	(501 x .001540)	=	4.00
6.31	+	(501 x .002098)	=	7.37
9.39	+	(501 x .002650)	=	10.72
12.47	+	(501 x .003150)	=	14.06
15.52	+	(501 x .00350)	=	17.29
18.49	+	(501 x .003282)	=	20.14
21.41	+	(501 x .00277)	=	22.8

Authority: A.D.M. 1061, page 17.

# DETERMINATION OF CRITICAL ANGLES OF ATTACK:

Equivalent monoplane aspect ratio 4.22

AR	AR model	AR model	AR model
4.22	4.22	4.22	4.22
1.2	1.2	1.2	1.2
1.4	1.4	1.4	1.4
1.6	1.6	1.6	1.6
1.8	1.8	1.8	1.8
2.0	2.0	2.0	2.0
2.2	2.2	2.2	2.2
2.4	2.4	2.4	2.4
2.6	2.6	2.6	2.6
2.8	2.8	2.8	2.8
3.0	3.0	3.0	3.0
3.2	3.2	3.2	3.2
3.4	3.4	3.4	3.4
3.6	3.6	3.6	3.6
3.8	3.8	3.8	3.8
4.0	4.0	4.0	4.0
4.2	4.2	4.2	4.2
4.4	4.4	4.4	4.4
4.6	4.6	4.6	4.6
4.8	4.8	4.8	4.8
5.0	5.0	5.0	5.0
5.2	5.2	5.2	5.2
5.4	5.4	5.4	5.4
5.6	5.6	5.6	5.6
5.8	5.8	5.8	5.8
6.0	6.0	6.0	6.0
6.2	6.2	6.2	6.2
6.4	6.4	6.4	6.4
6.6	6.6	6.6	6.6
6.8	6.8	6.8	6.8
7.0	7.0	7.0	7.0
7.2	7.2	7.2	7.2
7.4	7.4	7.4	7.4
7.6	7.6	7.6	7.6
7.8	7.8	7.8	7.8
8.0	8.0	8.0	8.0
8.2	8.2	8.2	8.2
8.4	8.4	8.4	8.4
8.6	8.6	8.6	8.6
8.8	8.8	8.8	8.8
9.0	9.0	9.0	9.0
9.2	9.2	9.2	9.2
9.4	9.4	9.4	9.4
9.6	9.6	9.6	9.6
9.8	9.8	9.8	9.8
10.0	10.0	10.0	10.0

# DETERMINATION OF $K_{m.c.g.}$

Cellule $\alpha$	$K_y$	Cellule $K_x$	$K_r$	$(\beta' - \alpha')$	Inches to C.P. from L.E.C.G. M.A.C.	Inches from vector to $K_{m.c.g.}$	
-6.1	-.0001536	.0000279	-.000155	43	65.5	83.5	-.0002205
-4.42	.0001152	.0000275	.000118	172	116.7	87.4	-.0001758
-2.72	.0004275	.0000329	.000429	7	42.25	19.4	-.000142
-1.05	.000686	.0000418	.000687	42	31.75	10.7	-.0001252
.65	.000983	.0000570	.000985	22	25.4	4.9	-.0000822
2.34	.001282	.0000803	.001284	12	21.6	1.5	-.0000328
4.00	.001540	.0001052	.001544	0	21.8	2.1	-.00005525
7.37	.002098	.000177	.002108	22	20.3	1.4	-.0000502
10.72	.00265	.000273	.002656	42	17.73	-0.4	.0000181
14.06	.00315	.000384	.003172	62	17.26	-0.4	.0000216
17.29	.00350	.000488	.003536	92	17.97	0.9	-.0000543
20.14	.003283	.000683	.003352	8	17.26	.0	0

$(\beta' - \alpha') = \text{Angle between normal to chord and resultant vector,}$   
 $K_r = \sqrt{K_y^2 + K_x^2}$

By extrapolation of  $K_{m.c.g.}$  curve:

-7.0	-.0002475
-8.0	-.000277
-9.0	-.000305
-10.0	-.000336

# DETERMINATION OF $K_m$ c.g.

Cellulose $K_y$	Cellulose $K_x$	$K_T$	( - )	Inches	Inches	from C.P. to M.V.C. vector	from I.E.C.G. to $K_m$ c.g.
-6.1	-0.001236	.0000273	-0.000122	43	62.5	83.5	-0.0002502
-4.42	.0001122	.0000272	.000118	173	116.7	87.4	-0.0001728
-2.72	.0004272	.0000223	.000422	7	42.22	19.4	-0.000142
-1.02	.000686	.0000418	.000687	43	21.72	10.7	-0.0001222
.62	.000282	.0000270	.000282	23	22.4	4.2	-0.0000822
2.24	.001222	.0000802	.001224	13	21.6	1.2	-0.0000222
4.00	.001240	.0001022	.001244	0	21.8	2.1	-0.0000222
7.27	.002028	.000177	.002108	24	20.3	1.4	-0.0000202
10.72	.00262	.000272	.002626	43	17.72	-0.4	.0000181
14.06	.00312	.000284	.003172	63	17.26	-0.4	.0000216
17.22	.00320	.000488	.003226	24	17.27	0.2	-0.0000242
20.14	.003222	.000682	.003222	8	17.26	0	0

( - ) Angle between normal to chord and resultant vector,  $K_T$   $K_y$   $K_x$

By extrapolation of  $K_m$  c.g. curve:

-10.0	-0.00326
-9.0	-0.00302
-8.0	-0.00277
-7.0	-0.002472

DETERMINATION OF AIRPLANE  $K_y$  :

$$\text{Airplane } K_y = \text{Cellule } K_y + \frac{\text{Tail Load}}{A_w V^2}$$

$$\frac{\text{Tail Load}}{A_w V^2} = K_{mc.g.} \times \frac{A_w V^2 \times C}{A_w V^2} = K_{mc.g.} \times \frac{58.7}{167.9} = .3495 K_{mc.g.}$$

Cellule

$\alpha$	(Cellule $K_y$ + .3495 $K_{mc.g.}$ )	= Airplane $K_y$
-6.10	(-.0001536 - .3495 x .0002205)	= -.0002307
-4.42	(.0001152 - .3495 x .0001758)	= .0000537
-2.72	(.0004275 - .3495 x .000142)	= .000378
-1.05	(.000686 - .3495 x .000110)	= .000648
.65	(.000983 - .3495 x .0000822)	= .0009543
2.34	(.001282 - .3495 x .000055)	= .001263
4.00	(.001540 - .3495 x .000040)	= .001526
7.37	(.002098 - .3495 x .000015)	= .002093
10.72	(.00265 - .3495 x .000002)	= .00265
14.06	(.00315 - .3495 x .000002)	= .00315
17.29	(.00350 - .3495 x .000020)	= .00350
20.14	(.003283 - .3495 x .00005)	= .003266

Note: Values of  $K_{mc.g.}$  above, taken from faired curve.

By extrapolation of Airplane  $K_y$  curve.

-7.0	-.00041
-8.0	-.00059
-9.0	-.00077
-10.0	-.00095

DETERMINATION OF AIRPLANE  $K_V$

$$\frac{W}{V} \times \frac{V}{W} \times \frac{C}{B} \times \text{Kmc.E.} = \frac{W}{V} \times \frac{V}{W} \times \frac{C}{B} \times \frac{167.9}{58.7} \times \text{Kmc.E.}$$

Airplane  $K_V$       Cellule  $K_V$       Tail Load

Cellule	(Cellule $K_V$ .3495 Kmc.E.)	Airplane $K_V$
20.14	(.003283 - .3495 x .00002)	.003266
17.22	(.003280 - .3495 x .000020)	.003250
14.06	(.003275 - .3495 x .000022)	.003235
10.72	(.003265 - .3495 x .000025)	.003225
7.37	(.003258 - .3495 x .000028)	.003218
4.00	(.003240 - .3495 x .000040)	.003200
2.34	(.003232 - .3495 x .000052)	.003192
.62	(.003223 - .3495 x .000082)	.003183
-1.05	(.003216 - .3495 x .000110)	.003176
-2.72	(.003207 - .3495 x .000142)	.003167
-4.42	(.003197 - .3495 x .000178)	.003157
-6.10	(.003186 - .3495 x .000202)	.003146

Note: Values of Kmc.E. above, taken from fitted curve.

By extrapolation of Airplane  $K_V$  curve.

-10.0	.00312
-2.0	.00317
-8.0	.00322
-7.0	.00327

DETERMINATION OF AIRPLANE  $K_X$  :

$$\text{Airplane } K_X = K_X(\text{cellule}) + \frac{A_e}{A_w} \times .00327 = K_X(\text{cellule}) + \frac{8.28}{252} \times .00327$$

$$= K_X(\text{cellule}) + .0001075$$

Cellule  $\alpha$        $K_X(\text{cellule}) + .0001075 = K_X$  (airplane.

-6.1	.0000279	+	.0001075	=	.0001354
-4.42	.0000275	+	.0001075	=	.0001350
-2.72	.0000329	+	.0001075	=	.0001404
-1.05	.0000418	+	.0001075	=	.0001493
.65	.0000570	+	.0001075	=	.0001645
2.34	.0000803	+	.0001075	=	.0001878
4.00	.0001052	+	.0001075	=	.0002127
7.37	.000177	+	.0001075	=	.0002845
10.72	.000273	+	.0001075	=	.0003805
14.06	.000384	+	.0001075	=	.0004915
17.29	.000488	+	.0001075	=	.0005955
20.14	.000683	+	.0001075	=	.0007905

By extrapolation of Airplane  $K_X$  curve.

-7.0	.000136
-8.0	.000137
-9.0	.000138
-10.0	.000142

# DETERMINATION OF AIRPLANE K<sub>x</sub> :

$$\text{Airplane } K_x(\text{celmle}) - \frac{A_s}{A_w} \times .00327 K_x(\text{celmle}) + \frac{8.58}{528} \times .00327 K_x(\text{celmle}) = .0001075$$

Celmle	K <sub>x</sub> (celmle)	.0001075	K <sub>x</sub> (airplane)
-6.1	.0000275	.0001075	.0001354
-4.42	.0000275	.0001075	.0001350
-2.72	.0000275	.0001075	.0001404
-1.02	.0000418	.0001075	.0001433
.62	.0000270	.0001075	.0001645
2.34	.0000802	.0001075	.0001878
4.00	.0001022	.0001075	.0002127
7.37	.000177	.0001075	.0002845
10.72	.000273	.0001075	.0003802
14.06	.000384	.0001075	.0004912
17.22	.000488	.0001075	.0005252
20.14	.000682	.0001075	.0007202

By extrapolation of Airplane K<sub>x</sub> curve.

-10.0	.000142
-9.0	.000138
-8.0	.000137
-7.0	.000136



DETERMINATION OF VELOCITIES:

$$V = \sqrt{\frac{W}{A}} \sqrt{\frac{\cos \beta}{K_y}} = \sqrt{\frac{2580}{252}} \sqrt{\frac{\cos \beta}{K_y}} = 3.2 \sqrt{\frac{\cos \beta}{K_y}} \quad \beta = \text{Flight path angle.}$$

$$-3.2 \sqrt{\frac{\sin \beta}{K_x}}$$

Cellule $\alpha$	Airplane $\frac{L}{D}$	Flight Path angle $\beta = \cot^{-1} \frac{L}{D}$	$\cos \beta$	Airplane $K_y$	$\frac{\cos \beta}{K_y}$	V
-6.1	-1.70	(-) 30°-28'	-.8619	-.0002307	3740	196
-4.7	0	90°	0	1.000	7400	275
-4.42	.398	68°-18'	.3697	.0000537	6880	266
-2.72	2.69	20°-24'	.9373	.000378	2480	159.5
-1.05	4.34	12°-59'	.9744	.000648	1504	124
.65	5.8	9°-47'	.9855	.0009543	1034	101.5
2.34	6.72	8°-28'	.9891	.001263	783	89.6
4.00	7.17	7°-56'	.9904	.001526	649	81.5
7.37	7.36	7°-44'	.9909	.002093	473	69.6
10.72	6.96	8°-11'	.9898	.00265	373	61.8
14.06	6.40	8°-53'	.9880	.00315	314	56.7
17.29	5.88	9°-39'	.9859	.00350	282	53.7
20.14	4.13	13°-37'	.9719	.003266	297	55.2

By extrapolation of Airplane characteristic curves:

-7	-3.01	(-) 18°-23'	-.9490	-.00041	2310	154
-8	-4.3	(-) 13°-05'	-.9740	-.00059	1650	130
-9	-5.58	(-) 10°-09'	-.9843	-.00077	1280	114.5
-10	-6.6	(-) 8°-37'	-.9887	-.00095	1040	103

DETERMINATION OF VELOCITIES

[illegible]

By extrapolation of Airplane characteristic curves:

451	0153	14000.-	0040.-	83-81 (-)	10.3-	7-
051	1030	00000.-	0470.-	02-81 (-)	3.4-	8-
2.41	0831	77000.-	3430.-	00-01 (-)	83.3-	0-
301	0401	00000.-	7880.-	78-8 (-)	0.0-	01-

EFFECT OF C.G. LOCATION ON VALUE  $K_M$  c.g. AT ZERO LIFT:

(4)

C.G. at ( 25% M.A.C.  
( On M.A.C.

Cellule $\alpha$	Inches from C.G. to vector	$K_r$	$K_M$ c.g.
-7.0	38.7	-.000326	-.000215
-6.08	79.6	-.000156	-.000212
-4.44	101.0	.000118	-.000203
-2.78	27.2	.000429	-.000199
-1.15	17.3	.000687	-.0002025

$K_M$  c.g. at zero lift = -.000205

EFFECT OF C.G. LOCATION ON VALUE K c.g. AT ZERO LIFT:

(4)

C.G. at ( ) ON M.A.C.  
( ) SSS M.A.C.

Celcius	Inches from C.G. to vector	K <sub>r</sub>	K <sub>c.g.</sub>
-7.0	38.7	--.000356	--.000315
-6.08	72.6	--.000156	--.000315
-4.44	101.0	.000118	--.000303
-3.78	57.2	.000423	--.000133
-1.15	17.3	.000687	--.000303
K <sub>c.g.</sub> at zero lift --.000303			

EFFECT ON VALUE OF NORMAL TAIL LOAD OF USING AIRFOIL  $K_M$   
AT ZERO LIFT FOR  $K_M$  c.g. AT VARIOUS C.G. LOCATIONS.

CLARK "Y" ,  $K_M$  at zero lift =  $-.00021$

(5) C.G. at 34.1% M.A.C.

$$\text{Normal Tail Load} = \frac{58.7}{167.5} \times 252 \times 82000 \times -.00021 = 1520$$

(6) C.G. at 25% M.A.C.

$$\text{Normal Tail Load} = 58.7 \times 252 \times 82000 \times -.00021 = 1475$$

(7) C.G. at 40% M.A.C.

$$\text{Normal Tail Load} = \frac{58.7}{164.} \times 252 \times 82000 \times -.00021 = 1550$$

EFFECT ON VALUE OF NORMAL TAIL LOAD OF USING AIRFOIL K  
AT ZERO LIFT FOR K M.C. AT VARIOUS C.G. LOCATIONS.

CLARK "Y" , K at zero lift 0.0001

(2) C.G. at 34.1% M.A.C.

Normal Tail Load  $58.7 \times 528 \times 82000 \times -.0001$  1250  
 $\underline{167.5}$

(3) C.G. at 32% M.A.C.

Normal Tail Load  $58.7 \times 528 \times 82000 \times -.0001$  1475

(7) C.G. at 40% M.A.C.

Normal Tail Load  $58.7 \times 528 \times 82000 \times -.0001$  1250  
 $\underline{164.}$

DETERMINATION OF TAIL LOADS:

$$\text{Normal Tail Load} = K_{M_{c.g.}} \times A_w \times \frac{C}{d} \times V^2 = K_{M_{c.g.}} \times \frac{252 \times 58.7}{167.9} \times V^2$$

$$= 88.1 \times V^2 \times K_{M_{c.g.}}$$

$$\text{Dynamic Tail Load} = \text{Normal Tail Load} \times \left( \frac{V_{\max}}{V} \right)^2$$

$\alpha$	$K_{M_{c.g.}}$	$V^2$	Normal Tail Load	$\left( \frac{V_{\max}}{V} \right)^2$	Dynamic Tail Load.
-6.1	-.0002205	38416	-746	1.97	1470
-4.7	-.000185	75600	-1232	1.00	1232
-4.42	-.0001758	70750	-1095	1.07	1172
-2.72	-.000142	25450	-318	2.97	945
-1.05	-.000110	15376	-148.7	4.92	731
0.65	-.0000822	10300	-74.5	7.56	563
2.34	-.000055	8028	-38.8	9.43	366
4.00	-.000040	6642	-23.4	11.4	267
7.37	-.000015	4844	-6.4	15.6	99.8
10.72	-.000002	3819	-0.7	19.8	13.3
14.06	-.000002	3215	-0.6	23.5	13.3
17.29	-.000020	2884	-5.1	26.2	133
20.14	-.000050	3047	-13.4	24.8	332

By extrapolation of Airplane characteristic curves:

-7.0	-.0002475	23650	-516	3.18	1640
-8.0	-.000277	16900	-412	4.47	1840
-9.0	-.000305	13220	-355	5.76	2045
-10.0	-.000336	10609	-314	7.12	2235

# DETERMINATION OF TAIL LOADS:

$$\text{Normal Tail Load } K_{No. 2} \times W \times C \times V \times K_{No. 2} \times \frac{167.9}{25258.7} \times V$$

$$88.1 \times V \times K_{No. 2}$$

$$\text{Dynamic Tail Load } \text{Normal Tail Load} \times \frac{V_{max}}{V}$$

Dynamic Tail Load	$\frac{V_{max}}{V}$	Normal Tail Load	V	$K_{No. 2}$	
1470	1.97	-746	38416	-.0002805	-6.1
1235	1.00	-1235	72600	-.000182	-4.7
1175	1.07	-1095	70750	-.0001728	-4.45
945	2.97	-318	32450	-.000142	-2.72
731	4.92	-148.7	12376	-.000110	-1.02
563	7.56	-74.2	10300	-.0000825	0.62
366	9.43	-38.8	8028	-.000022	2.34
267	11.4	-23.4	6642	-.000040	4.00
99.8	12.6	-6.4	4844	-.000012	7.37
13.3	12.8	-0.7	3812	-.000002	10.72
13.3	23.2	-0.6	3212	-.000002	14.06
133	26.2	-2.1	2884	-.000020	17.22
332	24.8	-12.4	2047	-.000020	20.14

By extrapolation of Airplane characteristic curves:

1640	2.18	-216	23650	-.0002472	-7.0
1840	4.47	-412	16300	-.000277	-8.0
2045	2.76	-322	13220	-.000302	-9.0
2232	7.12	-214	10602	-.000326	-10.0



EFFECT OF C.G. LOCATION ON VALUE OF NORMAL TAIL LOAD.AT ZERO LIFT:

$$\text{Normal Tail Load} = K_M \text{ c.g.} \times A_w \times \frac{C}{d} \times V^2$$

- (1) C.G. at { 40% M.A.C.  
                  { 25% below M.A.C.

$$d = 167.5 - (.40 - .341) \times 58.7 = (167.5 - 3.46) = 164"$$

$$\text{Normal Tail Load} = K_M \text{ c.g.} \times 252 \times \frac{58.7}{164} \times V^2 = 90.15 \times V^2 \times K_M \text{ c.g.}$$

Cellule $\alpha$	( 90.15 x $V^2$ x $K_M$ c.g.)	= Normal Tail Load.
-7.0	( 90.15 x 25770 x -.0002695)	= -625.
-6.08	( 90.15 x 41350 x -.00023 )	= -857.2
-4.7	( 90.15 x 82000 x -.000178 )	= -1315
-4.44	( 90.15 x 76700 x -.00017 )	= -1176
-2.78	( 90.15 x 27550 x -.000111 )	= - 276
-1.15	( 90.15 x 16800 x -.0000737 )	= - 111.7

- (2) C.G. at { 40% M.A.C.  
                  { On M.A.C.

$$d = 164"$$

$$\text{Normal Tail Load} = 90.15 \times V^2 \times K_M \text{ c.g.}$$

Cellule $\alpha$		
-7.0	( 90.15 x 25770 x -.000264)	= - 612.5
-6.08	( 90.15 x 41350 x -.000235)	= - 876
-4.75	( 90.15 x 82000 x -.000193)	= -1426
-4.44	( 90.15 x 76700 x -.000186)	= -1286
-2.78	( 90.15 x 27550 x -.0001353)	= - 336
-1.15	( 90.15 x 16800 x -.0000984)	= - 149

EFFECT OF C.G. LOCATION ON VALUE OF NORMAL TAIL LOAD.

AT ZERO LIFT:

$$\text{Normal Tail Load } K M c.g. = W \times \frac{C}{L} \times V$$

(1) C.G. at ( 40% M.A.C. )  
( 52% below M.A.C. )

b 167.5 - ( 40-.341 ) x 58.7 ( 167.5 - 3.46 ) 164"

$$\text{Normal Tail Load } K M c.g. = \frac{58.7}{164} \times 50.15 \times V \quad K M c.g.$$

Celcius ( 50.15 x V x K M c.g. ) Normal Tail Load.

-1.15	( 50.15 x 16800 x -.0000737	-652.
-2.78	( 50.15 x 57250 x -.000111	- 276
-4.44	( 50.15 x 76700 x -.00017	-1176
-4.7	( 50.15 x 82000 x -.000178	-1315
-6.08	( 50.15 x 41350 x -.00023	-827.5
-7.0	( 50.15 x 23770 x -.0002625	-652.

(2) C.G. at ( 40% M.A.C. )  
( On M.A.C. )

b 164"

$$\text{Normal Tail Load } 50.15 \times V \times K M c.g.$$

Celcius

-1.15	( 50.15 x 16800 x -.000084	- 148
-2.78	( 50.15 x 57250 x -.000123	- 336
-4.44	( 50.15 x 76700 x -.000186	-1286
-4.75	( 50.15 x 82000 x -.000193	-1426
-6.08	( 50.15 x 41350 x -.00023	- 876
-7.0	( 50.15 x 23770 x -.000264	- 612.5

COMPARISON OF AIRFOIL MOMENT COEFFICIENT AT ZERO LIFT  
AND  $K_M$  c.g. AT ZERO LIFT FOR SEVERAL AIRPLANES:

F6C-4 Airplane:

Clark "Y" airfoil.

$$K_{M0} = -.00021$$

( N.A.C.A.) (M.I.T.)

$$K_M \text{ c.g.} = -.0001875$$

(Fig. 12)

PW-9 Airplane:

Gott. 436 airfoil.

$$K_{M0} = -.00020$$

(M.I.T.)

$$K_M \text{ c.g.} = -.000189$$

( A.D.M. 1061)

B-2 Airplane:

C-72 airfoil.

$$K_{M0} = -.000246$$

( A.D.M. 1061)

$$K_M \text{ c.g.} = -.000235$$

( A.D.M. 1061)

A-3 Airplane:

Clark "Y" airfoil.

$$K_{M0} = -.00021$$

( N.A.C.A.)

$$K_M \text{ c.g.} = -.000172$$

( A.D.M. 1061)

PT-3A Airplane:

Clark "Y" Airfoil.

$$K_{M0} = -.00021$$

( N.A.C.A.)

$$K_M \text{ c.g.} = -.00021$$

( A.D.M. 1061)

COMPARISON OF AIRFOIL MOMENT COEFFICIENT AT ZERO LIFT  
AND K<sub>0</sub> c.g. AT ZERO LIFT FOR SEVERAL AIRPLANES:

F6C-4 Airplane:

Clark "Y" airfoil.

K<sub>Mo</sub> --.00021

( M.A.C.A. ) ( M.I.T. )

K<sub>0</sub> c.g. --.0001875

( P.E. )

PW-3 Airplane:

Gott. 436 airfoil.

K<sub>Mo</sub> --.00020

( M.I.T. )

K<sub>0</sub> c.g. --.000189

( A.D.M. 1961 )

B-3 Airplane:

C-73 airfoil.

K<sub>Mo</sub> --.000246

( A.D.M. 1961 )

K<sub>0</sub> c.g. --.000235

( A.D.M. 1961 )

A-3 Airplane:

Clark "Y" airfoil.

K<sub>Mo</sub> --.00021

( M.A.C.A. )

K<sub>0</sub> c.g. --.000175

( A.D.M. 1961 )

PT-3A Airplane:

Clark "Y" airfoil.

K<sub>Mo</sub> --.00021

( M.A.C.A. )

K<sub>0</sub> c.g. --.00021

( A.D.M. 1961 )

EFFECT OF C.G. LOCATION ON VALUE OF  $K_{M\text{c.g.}}$  AT ZERO LIFT:

(1)

C.G. at { 40% M.A.C.  
45% below M.A.C.

Cellule $\alpha$	Inches from C.G. to vector	$K_F$	$K_{M\text{c.g.}}$
-7.0	48.5	-.000326	-.0002695
-6.08	86.5	-.000156	-.00023
-4.44	84.5	.000118	-.00017
-2.78	15.2	.000429	-.000111
-1.15	6.8	.000687	-.0000737

$K_{M\text{c.g.}}$  at zero lift = -.000178

(2)

C.G. at { 40% M.A.C.  
on M.A.C.

Cellule	Inches from C.G. to vector	$K_F$	$K_{M\text{c.g.}}$
-7.0	47.5	-.000326	-.000264
-6.08	88.4	-.000156	-.000235
-4.44	92.6	.000118	-.000186
-2.78	18.8	.000429	-.0001353
-1.15	8.4	.000687	-.0000984

$K_{M\text{c.g.}}$  at zero lift = -.000193

(3)

C.G. at (25% M.A.C.  
(45% below M.A.C.

Cellule	Inches from C.G. to vector	$K_F$	$K_{M\text{c.g.}}$
-7.0	39.7	-.000326	-.0002205
-6.08	77.7	-.000156	-.0002065
-4.44	92.7	.000118	-.0001864
-2.78	23.9	.000429	-.000175
-1.15	15.1	.000687	-.000177

$K_{M\text{c.g.}}$  at zero lift = -.000189

EFFECT OF C.G. LOCATION ON VALUE OF  $K_{c.g.}$  AT ZERO LIFT:

(1)

C.G. at ( 40% M.A.C. )  
45% below M.A.C.

Celulose	Inches from C.G. to vector	$K_T$	$K_{M.C.G.}$
-1.15	6.5	.000687	.0000737
-2.78	15.2	.000429	-.000111
-4.44	24.2	.000118	-.00017
-6.08	36.2	-.000126	-.00023
-7.0	48.2	-.000326	-.000262
KM c.g. at zero lift			
-.000178			

(2)

C.G. at ( 40% M.A.C. )  
( on M.A.C. )

Celulose	Inches from C.G. to vector	$K_T$	$K_{M.C.G.}$
-1.15	8.4	.000687	-.0000984
-2.78	18.2	.000429	-.0001323
-4.44	22.6	.000118	-.000186
-6.08	38.4	-.000126	-.000232
-7.0	47.2	-.000326	-.000264
KM c.g. at zero lift			
-.000123			

(3)

C.G. at ( 25% M.A.C. )  
( 45% below M.A.C. )

Celulose	Inches from C.G. to vector	$K_T$	$K_{M.C.G.}$
-1.15	12.1	.000687	-.000177
-2.78	23.2	.000429	-.000172
-4.44	32.7	.000118	-.0001864
-6.08	47.7	-.000126	-.000202
-7.0	52.7	-.000326	-.000202
KM c.g. at zero lift			
-.000182			

EFFECT OF C.G. LOCATION ON VALUE OF NORMAL TAILLOAD AT ZERO LIFT:

(3) C.G. at ( 25% M.A.C.  
( 45% below M.A.C.

$$d = 167.5 + (.341-.25) \times 58.7 = 167.5 + 5.29 = 172.8$$

$$\text{Normal Tail Load} = K_{M \text{ c.g.}} \times 252 \times \frac{58.7}{172.8} \times V^2 = 85.5 \times V^2 \times K_{M \text{ c.g.}}$$

Cellule $\alpha$	$(85.5 \times V^2 \times K_{M \text{ c.g.}})$	=	Normal Tail Load.
-7.0	$(85.5 \times 25770 \times -.0002205)$	=	- 486
-6.08	$(85.5 \times 41350 \times -.0002065)$	=	- 730
-4.7	$(85.5 \times 82000 \times -.000189)$	=	-1326
-4.44	$(85.5 \times 76700 \times -.0001865)$	=	-1224
-2.78	$(85.5 \times 27550 \times -.000175)$	=	- 412
-1.15	$(85.5 \times 16800 \times -.000177)$	=	- 254

(4) C.G. at (25% M.A.C.  
(On M.A.C.

$$d = 172.8''$$

-7.0	$(85.5 \times 25770 \times -.000215)$	=	- 474
-6.08	$(85.5 \times 41350 \times -.000212)$	=	- 750
-4.7	$(85.5 \times 82000 \times -.000205)$	=	-1438
-4.44	$(85.5 \times 76700 \times -.000203)$	=	-1330
-2.78	$(85.5 \times 27550 \times -.000199)$	=	- 469
-1.15	$(85.5 \times 16800 \times -.0002025)$	=	- 291





COMPARISON OF AIRFOIL MOMENT COEFFICIENT AT ZERO LIFT  
AND  $K_M$  c.g. AT ZERO LIFT, FOR SEVERAL AIRPLANES:

C-2 Airplane:

Fokker airfoil.

$K_{Mo}$  = Not recorded.

$K_M$  c.g. =  $-.0001614$  ( A.D.M. 1061)

LB-7 Airplane.

Gott. 398 airfoil.

$K_{Mo}$  =  $-.000228$  (W.H.Y.)

$K_M$  c.g. =  $-.00022$  ( A.D.M. 1061)

COMPARISON OF AIRFOIL MOMENT COEFFICIENT AT ZERO LIFT  
AND  $C_{m,0}$  AT ZERO LIFT, FOR SEVERAL AIRPLANES:

C-2 Airplane:

Fokker airplane.

$C_{m,0}$

(A.D.M. 1961)

$C_{m,0} = -.00014$

LB-7 Airplane.

Coff. 398 airplane.

(W.M.Y.)

$C_{m,0} = -.000328$

(A.D.M. 1961)

$C_{m,0} = -.00032$

COMPARATIVE TERMINAL VELOCITY CALCULATIONS:

Data for the calculations of terminal velocity from the drag formula, have been taken from A.D.M. 1061, except for the F6C-4 airplane. Data for the calculations of terminal velocity from the power absorbed formula, have been taken from A.C.I.C. #629, except for the case of the F6C-4 airplane.

F6C-4 Airplane:

$$W \cos \beta = K_x A_w V^2 \quad \beta = \text{flight path angle} = 90^\circ$$

$$(1) \quad V = \sqrt{\frac{W}{K_x A_w}} = \sqrt{\frac{2796}{.000135 \times 252}} = 286.5 \text{ m.p.h.}$$

$$(2) \quad V = \sqrt{\frac{V_{\max}^3 \times W}{375 \times \gamma_{\max} \times \text{B.H.P.}}} = \sqrt{\frac{(157)^3 \times 2796}{375 \times .84 \times 435}} = 281 \text{ m.p.h.}$$

PW-9 Airplane:

$$(1) \quad V = \sqrt{\frac{2890}{.0001227 \times 240.76}} = 312 \quad \text{A.D.M. 1061}$$

$$(2) \quad V = \sqrt{\frac{(165.5)^3 \times 2890}{375 \times .831 \times 431}} = 312$$

B-2 Airplane:

$$(1) \quad V = \sqrt{\frac{16610}{.000118 \times 1498}} = 306$$

$$(2) \quad V = \sqrt{\frac{(129.85)^3 \times 16610}{375 \times .816 \times 1227}} = 311$$

C-2 Airplane:

$$(1) \quad V = \sqrt{\frac{10393}{.00017 \times 747}} = 286$$

$$(2) \quad V = \sqrt{\frac{(116)^3 \times 10393}{375 \times .799 \times 671}} = 284$$

# COMPARATIVE TERMINAL VELOCITY CALCULATIONS:

Data for the calculations of terminal velocity from the drag formula, have been taken from A.D.M. 1061, except for the POC-4 airplane. Data for the calculations of terminal velocity from the power absorbed formula, have been taken from A.C.I.C. 4622, except for the case of the POC-4 airplane.

## POC-4 Airplane:

$$\begin{aligned} (1) \quad V &= \frac{K \cdot W}{K \cdot W} \quad \text{W cos } K \cdot W \quad \text{flight path angle } 90^\circ \\ &= \frac{2796}{.000135 \times 525} \quad 586.5 \text{ m.p.h.} \\ (2) \quad V &= \frac{V_{\text{max}} \times W}{3.75 \times \text{max } x \text{ R.H.P.}} \quad (157) \times 2796 \quad 581 \text{ m.p.h.} \\ &= \frac{3.75 \times .84 \times 435}{3.75 \times \text{max } x \text{ R.H.P.}} \end{aligned}$$

## PW-3 Airplane:

$$\begin{aligned} (1) \quad V &= \frac{2890}{.0001227 \times 240.76} \quad 315 \\ (2) \quad V &= \frac{(165.2) \times 2890}{3.75 \times .831 \times 431} \quad 315 \end{aligned}$$

## B-2 Airplane:

$$\begin{aligned} (1) \quad V &= \frac{16610}{.000118 \times 1498} \quad 306 \\ (2) \quad V &= \frac{(132.82) \times 16610}{3.75 \times .816 \times 1224} \quad 311 \end{aligned}$$

## C-2 Airplane:

$$\begin{aligned} (1) \quad V &= \frac{10393}{.00014 \times 444} \quad 386 \\ (2) \quad V &= \frac{(116) \times 10393}{3.75 \times .722 \times 641} \quad 384 \end{aligned}$$

COMPARATIVE TERMINAL VELOCITY CALCULATIONS: (cont'd)A-3 Airplane:

$$(1) \quad V = \sqrt{\frac{4377}{.0001338 \times 353}} = 304.5$$

$$(2) \quad V = \sqrt{\frac{(141.4)^3 \times 4377}{375 \times .806 \times 440}} = 305$$

PT-3A Airplane:

$$(1) \quad V = \sqrt{\frac{2431}{.000189 \times 300}} = 207$$

$$(2) \quad V = \sqrt{\frac{(105.5)^3 \times 2431}{375 \times .796 \times 220}} = 208$$

LB-7 Airplane:

$$(1) \quad V = \sqrt{\frac{12868}{.0001605 \times 1150}} = 264$$

$$(2) \quad V = \sqrt{\frac{(117.6)^3 \times 12868}{375 \times .784 \times 1070}} = 258$$

A-3 Airplane:

$$\begin{array}{r} 4377 \\ 804.8 \times 0.001338 \end{array} \quad V \quad (1)$$

$$\begin{array}{r} 4377 \times (4.141) \\ 804.8 \times 440 \end{array} \quad V \quad (2)$$

PT-3A Airplane:

$$\begin{array}{r} 2431 \\ 808 \times 0.00182 \end{array} \quad V \quad (1)$$

$$\begin{array}{r} 2431 \times (2.102) \\ 808 \times 220 \end{array} \quad V \quad (2)$$

LB-7 Airplane:

$$\begin{array}{r} 12868 \\ 828 \times 0.001602 \end{array} \quad V \quad (1)$$

$$\begin{array}{r} 12868 \times (2.711) \\ 828 \times 1070 \end{array} \quad V \quad (2)$$

COMPARATIVE MAXIMUM NORMAL TAIL LOAD CALCULATIONS:

(1) Formula: Max. Normal Tail Load =  $K_M \text{ c.g.} \times \frac{C}{d} \times A_w \times V_t^2$

$V_t$  from,  $W \cos \beta = K_x A_w V_t^2$

(2) Formula: Max. Normal Tail Load =  $K_{M_0} \times \frac{C}{d} \times A_w \times V_t^2$

where  $K_{M_0}$  = Airfoil  $K_M$  at zero lift,

and,  $V_t = \sqrt{\frac{V_{\max}^3 \times W}{375 \times \eta_{\max} \times \text{B.H.P.}}}$

F6C-4 Airplane:

(1) Max. Normal Tail Load = -1359 lbs. (page 47)

(2) Max. Normal Tail Load =  $-.00021 \times 58.7 \times 252 \times (281)^2 = -1465$   
167.5

PW-9 Airplane:

(1) Max. Normal Tail Load = -1480 (A.D.M. 1061)

(2) Max. Normal Tail Load =  $-.00020 \times 59.15 \times 240.76 \times (312)^2$   
177.3  
= -1565

B-2 Airplane:

(1) Max. Normal Tail Load = -9500 (A.D.M. 1061)

(2) Max. Normal Tail Load =  $-.000246 \times 108.8 \times 1498 \times (311)^2$   
377  
= -10300

A-3 Airplane:

(1) Max. Normal Tail Load = -1920 (A.D.M. 1061)

(2) Max. Normal Tail Load =  $-.00021 \times 64.3 \times 353 \times (305)^2 = -2220$   
200

COMPARATIVE MAXIMUM NORMAL TAIL LOAD CALCULATIONS:

(1) Formula: Max. Normal Tail Load  $K M_o \cdot \frac{C}{D} \times A_w \times V_f$

$V_f$  from,  $W \cos K \times A_w \times V_f$

(2) Formula: Max. Normal Tail Load  $K M_o \times \frac{C}{D} \times A_w \times V_f$

where  $K M_o$  Airfoil  $K M_o$  zero lift.

$$\text{and } V_f = \frac{V_{max} \times W}{375 \times B.H.P.}$$

F6C-4 Airplane:

(1) Max. Normal Tail Load -1352 lbs. (page )

(2) Max. Normal Tail Load -0.0051 x 58.7 x 525 x (381) -1462  
167.5

F7W-3 Airplane:

(1) Max. Normal Tail Load -1480 (A.D.M. 1061)

(2) Max. Normal Tail Load -0.0050 x 59.15 x 540.76 x (315)  
177.3

= -1565

B-3 Airplane:

(1) Max. Normal Tail Load - 9500 (A.D.M. 1061)

(2) Max. Normal Tail Load -0.00546 x 108.8 x 1498 x (311)  
377

= -10300

A-3 Airplane:

(1) Max. Normal Tail Load -1950 (A.D.M. 1061)

(2) Max. Normal Tail Load -0.0051 x 64.3 x 523 x (305) -3350  
500



COMPARATIVE MAXIMUM TAIL LOAD CALCULATIONS: (cont'd)PT-3A Airplane:

$$(1) \text{ Max. Normal Tail Load} = -680 \quad (\text{A.D.M. 1061})$$

$$(2) \text{ Max. Normal Tail Load} = -.00021 \times \frac{56}{222} \times 300 \times (208)^2 \\ = -687$$

C-2 Airplane:

$$(1) \text{ Max. Normal Tail Load} = -3160 \quad (\text{A.D.M. 1061})$$

$$(2) \text{ Max. Normal Tail Load} = \frac{128.9}{402} \times 747 \times (284)^2 \times K_M^* =$$

LB-7 Airplane:

$$(1) \text{ Max. Normal Tail Load} = -5000 \quad (\text{A.D.M. 1061})$$

$$(2) \text{ Max. Normal Tail Load} = -.000228 \times \frac{96.2}{340} \times 1150 \times (258)^2 \\ = -4940$$

\* Not recorded.

COMPARATIVE MAXIMUM TAIL LOAD CALCULATION: (cont'd)

FT-3A Airplane:

(1) Max. Normal Tail Load -680 (A.D.M. 1061)  
 (2) Max. Normal Tail Load --.00051 x 56 x 300 x (508)  
 $\frac{833}{}$   
 = -687

C-5 Airplane:

(1) Max. Normal Tail Load -3160 (A.D.M. 1061)  
 (2) Max. Normal Tail Load  $\frac{9.88.9}{405}$  x 747 x (584)

LB-7 Airplane:

(1) Max. Normal Tail Load -5000 (A.D.M. 1061)  
 (2) Max. Normal Tail Load --.00058 x 96.8 x 1150 x (528)  
 $\frac{340}{}$   
 = -4940

COMPARATIVE VALUES OF MAXIMUM NORMAL TAIL LOAD:

Air-plane	$K_M$ c.g. at zero lift.	Airfoil $K_M$ at zero lift.	$V_t$ (A.D.M. 1061)	$V_t$ by formula (2) below	Max. normal Tail Load by (A.D.M. 1061)	Max. Normal Tail Load by proposed modification
F6C-4	-.0001875	-.00021	286.5	281	1359	1465
PW-9	-.000189	-.00020	312	312	1480	1565
B-2	-.000235	-.000246	306	311	9500	10300
A-3	-.000182	-.00021	304.5	305	1920	2220
PT-3A	-.00021	-.00021	207	208	680	687
C-2	-.0001614	—	286	284	3160	—
LB-7	-.00022	-.000228	264	258	5000	4940

$$(1) \quad W \cos 90^\circ = K_X A_W V_t^2 \quad (\text{A.D.M. 1061})$$

$$(2) \quad V = \sqrt{\frac{V_{\max}^3 \times W}{375 \times \eta_{\max} \times \text{B.H.P.}}} \quad (\text{power absorbed formula})$$

COMPARATIVE VALUES OF MAXIMUM NORMAL TAIL LOAD:

Air- plane	K c.g. at zero lift.	Airfoil K at zero lift.	V <sub>f</sub> (A.D.M. load) (S) below	V <sub>f</sub> Max. normal tail lift load by propos- ed mod- ification	Air- plane	K c.g. at zero lift.	Airfoil K at zero lift.	V <sub>f</sub> (A.D.M. load) (S) below	V <sub>f</sub> Max. normal tail lift load by propos- ed mod- ification
FC-4	--.0001875	--.00051	386.5	381	1553	1465			
FW-3	--.000189	--.00050	315	315	1480	1565			
B-2	--.000535	--.000546	306	311	3500	10300			
A-3	--.000185	--.00051	304.5	305	1350	3350			
PT-3A	--.00051	--.00051	307	308	680	687			
C-2	--.0001614		386	384	3160				
LB-7	--.00055	--.000558	364	358	5000	4340			

(1)  $W \cos 30^\circ K_A V_f$

(A.D.M. load)

(2)  $V \frac{V_{max}^2}{375} \times B.H.P.$

(power absorbed formula)

BOEING PW-9 FIGHTING AIRPLANE  
TAIL LOAD COMPUTATIONS.

BOEING PW-3 FIGHTING AIRPLANE  
TAIL LOAD COMPUTATIONS.

BASIC DATA ON PW-9 AIRPLANE:

Gross Weight	2890 lbs.
Wing Section--	Cott., 436
M.A.C. (C)--	59.15"
Span , upper--	32.00'
Span , lower--	22.50'
Chord, upper, (average)--	5.13'
Chord, lower, (average)--	4.51'
Gap---	4.30'
Stagger, at leading edge of fuselage---	5.1°
C.G. % M.A.C.---	31.
C.G. % M.A.C. below---	35.
Area of horizontal tail surfaces--	30.3 sq.ft.
Total wing area---	240.75 sq.ft.
Distance from C.G. to tail post -(d)--	177.3"
Area of upper wing---	159.8 sq.ft.
Area of lower wing---	80.96 sq.ft.
Maximum speed at sea level--	165.5 M.P.H.
B.H.P.	431.0
Diameter of propeller---	8.67 ft.

Authority: Air Corps Information Circular No. 629

# BASIC DATA ON PW-3 AIRPLANE:

8.67 ft.	Diameter of propeller---
431.0	B.H.P.
165.5 M.P.H.	Maximum speed at sea level--
80.96 sq.ft.	Area of lower wing---
159.8 sq.ft.	Area of upper wing---
177.3"	Distance from C.G. to tail post - (d)---
240.75 sq.ft.	Total wing area---
30.3 sq.ft.	Area of horizontal tail surfaces--
35	C.G. & M.A.C. below---
31	C.G. & M.A.C. ---
5.1°	Stagger, at leading edge of fuselage---
4.30'	Gap---
4.51'	Chord, lower, (average)---
5.13'	Chord, upper, (average)---
22.50'	Span, lower---
32.00'	Span, upper---
29.15"	M.A.C. (C)---
436	Wing section---
2890 lbs.	Gross Weight

Coff.,

Authority: Army Air Corps Information Circular No. 629



## GOTT. 436 AIRFOIL CHARACTERISTICS:

$\alpha^\circ$	$C_L$	$C_D$	$C_m$	$K_y$	$K_x$	$K_m$
-8.9	-.236	.0437	-0.009	-.000604	.000112	-.000023
-6.0	-.050	.0144	-0.063	-.000128	.00003685	-.000161
-4.5	.050	.0130	-0.084	.000128	.0000333	-.000215
-3.0	.150	.0133	-0.107	.000384	.00003405	-.000274
-1.6	.246	.0159	-0.130	.000630	.0000407	-.000333
-0.1	.349	.0189	-0.154	.000894	.0000484	-.000394
1.3	.451	.0247	-0.182	.001153	.0000632	-.000466.
2.8	.548	.0294	-0.202	.001405	.0000753	-.000517
4.3	.647	.0382	-0.226	.001657	.0000978	-.000579
5.7	.751	.0488	-0.248	.001922	.000125	-.000635
8.7	.945	.0728	-0.301	.00242	.0001866	-.00077
11.6	1.120	.0999	-0.343	.00287	.000254	-.000878
14.6	1.204	.138	-0.365	.00308	.0003535	-.000935

Authority: N.A.C.A. Report No. 233. (Gottingen Tunnel Data).

Authority: W.A.C.A. Report No. 232. (Göttingen Tunnel Data).

km	Km	Km	cm	cm	cm	cm
14.6	1.204	1.28	-0.365	0.0808	0.003222	-0.00232
11.6	1.120	0.999	-0.343	0.0287	0.00224	-0.00278
8.7	0.945	0.0728	-0.301	0.0242	0.001866	-0.00077
5.7	0.751	0.0488	-0.248	0.0122	0.000722	-0.000622
4.3	0.647	0.0382	-0.226	0.01627	0.000978	-0.00029
2.8	0.548	0.0294	-0.202	0.01402	0.000722	-0.000217
1.3	0.421	0.0247	-0.182	0.01122	0.000622	-0.000466
-0.1	0.349	0.0189	-0.124	0.00894	0.000484	-0.000294
-1.6	0.246	0.0129	-0.130	0.00630	0.000407	-0.000322
-3.0	0.120	0.0132	-0.107	0.00384	0.0002402	-0.000274
-4.2	0.020	0.0130	-0.084	0.00128	0.0000322	-0.000212
-6.0	-0.020	0.0144	-0.062	0.00128	0.00003682	-0.000161
-8.9	-0.226	0.0427	-0.002	0.00604	0.00012	-0.000022

GOTT. 436 AIRFOIL CHARACTERISTICS:  
cm Km Km Km Km Km

DETERMINATION OF CORRECTED  $\frac{L}{D}$  FOR CELLULE:

$$\text{Corrected } \frac{L}{D} \text{ cellule} = \frac{1}{\frac{D_{\text{model}}}{L} + B \times K_y}$$

$$B = \left( \frac{C_l^2 + C_u^2 + 2s C_l C_u}{A} - \frac{1}{AR_{\text{model}}} \right)$$

$$A = (\text{Total wing area} + \text{area under fuselage}) = 252.76 \text{ sq.ft.}$$

$$B = 125 \left( \frac{4.51^2 + 5.13^2 + 2 \times .48 \times 4.51 \times 5.13}{252.76} - \frac{1}{5} \right)$$

$$= 9.00$$

$$\frac{\text{Gap}}{\text{Mean span}} = \frac{4.30}{27.25} = 15.75$$

$$\frac{\text{Lower span}}{\text{upper span}} = \frac{22.5}{32.0} = .703$$

$$s = .48$$

Authority: Page 78, Instructions for Airplane Designers.

DETERMINATION OF CORRECTED I FOR CELLULE:

$$\text{Corrected } I_{\text{cellule}} = \frac{I}{\frac{I_{\text{model}}}{I} + B \times K \sqrt{A}}$$

$$B = \left( \frac{C_L + C_D}{A} + \frac{C_L}{S_2} + \frac{C_D}{S_1} \right) - \frac{I}{A R_{\text{model}}} \quad (1)$$

$$A = \text{Total wing area - area under fuselage} = 525.76 \text{ sq. ft.}$$

$$B = \frac{125 (4.51 + 5.13 \times 2 \times 48 \times 4.51 \times 5.13 - \frac{1}{2} \times 525.76)}{525.76} = 125$$

$$= 9.00$$

$\frac{\text{Gap}}{\text{Mean span}} = \frac{4.30}{27.25} = 15.75$	$=$	$15.75$
$\frac{\text{Lower span}}{\text{Upper span}} = \frac{22.5}{32.0} = 70.3$	$=$	$70.3$

$$= 48$$

Authority: Page 78, Instructions for Airplane Designers.

DETERMINATION OF CORRECTED  $\frac{L}{D}$  FOR CELLULE: (Cont'd)

Airfoil $\alpha^\circ$	Airfoil $K_y$	$\frac{1}{\frac{D}{L} \text{ model} + B \times K_y}$	$= \frac{L}{D}$ Corrected (cellule)
-8.9	-.000604		-5.25
-6.0	-.000128		-3.485
-4.5	.000128		3.85
-3.0	.000384		10.88
-1.6	.000630		14.25
-0.1	.000894		16.12
1.3	.001153		15.40
2.8	.001405		15.05
4.3	.001657		13.50
5.7	.001922		12.16
8.7	.00242		10.13
11.6	.00287		8.68
14.6	.00308		7.025

DETERMINATION OF CELLULE ANGLES OF ATTACK:

Equivalent monoplane aspect ratio 4.16

$$\alpha^\circ_{AR} = \alpha^\circ_{AR \text{ model}} - (125 \times 57.3 K_y) \left( \frac{1}{AR \text{ model}} - \frac{1}{AR} \right)$$

$$\alpha^\circ_{4.16} = -8.9 - (125 \times 57.3 \times -.000604) \left( \frac{1}{5} - \frac{1}{4.16} \right) = -9.07$$

-6.0	-6.04
-4.5	-4.46
-3.0	-2.89
-1.6	-1.415
-0.1	.158
1.3	1.63
2.8	3.205
4.3	4.775
5.7	6.25
8.7	9.395
11.6	12.43
14.6	15.49

# DETERMINATION OF CORRECTED $\frac{I}{D}$ FOR CELLULOSE (Cont'd)

Corrected (cellulose)	$\frac{I}{D}$	$\frac{I}{D}$ model $\cdot B \times K \lambda$	$K \lambda$	Actual	Actual
7.025	8.68	10.13	0.0122	11.6	14.6
8.68	10.13	12.16	0.0122	11.6	14.6
10.13	12.16	13.50	0.0122	11.6	14.6
12.16	13.50	15.05	0.0122	11.6	14.6
13.50	15.05	16.40	0.0122	11.6	14.6
15.05	16.40	17.13	0.0122	11.6	14.6
16.40	17.13	18.25	0.0122	11.6	14.6
17.13	18.25	19.88	0.0122	11.6	14.6
18.25	19.88	20.82	0.0122	11.6	14.6
20.82	20.82	21.28	0.0122	11.6	14.6
21.28	21.28	22.04	0.0122	11.6	14.6
22.04	22.04	23.00	0.0122	11.6	14.6
23.00	23.00	24.00	0.0122	11.6	14.6
24.00	24.00	25.00	0.0122	11.6	14.6
25.00	25.00	26.00	0.0122	11.6	14.6
26.00	26.00	27.00	0.0122	11.6	14.6
27.00	27.00	28.00	0.0122	11.6	14.6
28.00	28.00	29.00	0.0122	11.6	14.6
29.00	29.00	30.00	0.0122	11.6	14.6
30.00	30.00	31.00	0.0122	11.6	14.6
31.00	31.00	32.00	0.0122	11.6	14.6
32.00	32.00	33.00	0.0122	11.6	14.6
33.00	33.00	34.00	0.0122	11.6	14.6
34.00	34.00	35.00	0.0122	11.6	14.6
35.00	35.00	36.00	0.0122	11.6	14.6
36.00	36.00	37.00	0.0122	11.6	14.6
37.00	37.00	38.00	0.0122	11.6	14.6
38.00	38.00	39.00	0.0122	11.6	14.6
39.00	39.00	40.00	0.0122	11.6	14.6
40.00	40.00	41.00	0.0122	11.6	14.6
41.00	41.00	42.00	0.0122	11.6	14.6
42.00	42.00	43.00	0.0122	11.6	14.6
43.00	43.00	44.00	0.0122	11.6	14.6
44.00	44.00	45.00	0.0122	11.6	14.6
45.00	45.00	46.00	0.0122	11.6	14.6
46.00	46.00	47.00	0.0122	11.6	14.6
47.00	47.00	48.00	0.0122	11.6	14.6
48.00	48.00	49.00	0.0122	11.6	14.6
49.00	49.00	50.00	0.0122	11.6	14.6
50.00	50.00	51.00	0.0122	11.6	14.6
51.00	51.00	52.00	0.0122	11.6	14.6
52.00	52.00	53.00	0.0122	11.6	14.6
53.00	53.00	54.00	0.0122	11.6	14.6
54.00	54.00	55.00	0.0122	11.6	14.6
55.00	55.00	56.00	0.0122	11.6	14.6
56.00	56.00	57.00	0.0122	11.6	14.6
57.00	57.00	58.00	0.0122	11.6	14.6
58.00	58.00	59.00	0.0122	11.6	14.6
59.00	59.00	60.00	0.0122	11.6	14.6
60.00	60.00	61.00	0.0122	11.6	14.6
61.00	61.00	62.00	0.0122	11.6	14.6
62.00	62.00	63.00	0.0122	11.6	14.6
63.00	63.00	64.00	0.0122	11.6	14.6
64.00	64.00	65.00	0.0122	11.6	14.6
65.00	65.00	66.00	0.0122	11.6	14.6
66.00	66.00	67.00	0.0122	11.6	14.6
67.00	67.00	68.00	0.0122	11.6	14.6
68.00	68.00	69.00	0.0122	11.6	14.6
69.00	69.00	70.00	0.0122	11.6	14.6
70.00	70.00	71.00	0.0122	11.6	14.6
71.00	71.00	72.00	0.0122	11.6	14.6
72.00	72.00	73.00	0.0122	11.6	14.6
73.00	73.00	74.00	0.0122	11.6	14.6
74.00	74.00	75.00	0.0122	11.6	14.6
75.00	75.00	76.00	0.0122	11.6	14.6
76.00	76.00	77.00	0.0122	11.6	14.6
77.00	77.00	78.00	0.0122	11.6	14.6
78.00	78.00	79.00	0.0122	11.6	14.6
79.00	79.00	80.00	0.0122	11.6	14.6
80.00	80.00	81.00	0.0122	11.6	14.6
81.00	81.00	82.00	0.0122	11.6	14.6
82.00	82.00	83.00	0.0122	11.6	14.6
83.00	83.00	84.00	0.0122	11.6	14.6
84.00	84.00	85.00	0.0122	11.6	14.6
85.00	85.00	86.00	0.0122	11.6	14.6
86.00	86.00	87.00	0.0122	11.6	14.6
87.00	87.00	88.00	0.0122	11.6	14.6
88.00	88.00	89.00	0.0122	11.6	14.6
89.00	89.00	90.00	0.0122	11.6	14.6
90.00	90.00	91.00	0.0122	11.6	14.6
91.00	91.00	92.00	0.0122	11.6	14.6
92.00	92.00	93.00	0.0122	11.6	14.6
93.00	93.00	94.00	0.0122	11.6	14.6
94.00	94.00	95.00	0.0122	11.6	14.6
95.00	95.00	96.00	0.0122	11.6	14.6
96.00	96.00	97.00	0.0122	11.6	14.6
97.00	97.00	98.00	0.0122	11.6	14.6
98.00	98.00	99.00	0.0122	11.6	14.6
99.00	99.00	100.00	0.0122	11.6	14.6

## DETERMINATION OF CELLULOSE ANGLES OF ATTACK:

Equivalent monolayer aspect ratio 4.16

$$AR = AR \text{ model} - (125 \times 27.3 K \lambda) \left( \frac{I}{AR \text{ model}} \right) \left( \frac{I}{AR} \right) - 9.07$$

$$4.16 = 8.2 (125 \times 27.3 \times -0.00604) \left( \frac{I}{4.16} \right) - 9.07$$

12.42	14.6
15.43	11.6
23.32	8.7
23.32	7.2
4.77	4.3
3.20	2.8
1.63	1.3
1.28	-0.1
-1.41	-1.6
-2.89	-3.0
-4.46	-4.2
-6.04	-6.0

DETERMINATION OF  $K_M$  e.g. :

Cellule $\alpha$	$(\beta^\circ - \alpha^\circ)$	Inches from L.E. to C.P.	Inches from C.G. to vector.	$K_M$ e.g.
-9.07	↓ 1.5	2.21	20.0	-.000208
-6.04	↓ 10.0	72.15	85.8	-.000196
-4.46	19.0 ↑	101.8	70.7	-.000158
-2.89	8.0 ↑	42.5	21.0	-.000137
-1.42	5.25 ↑	31.4	10.8	-.0001152
.16	3.5 ↑	26.0	6.25	-.0000946
1.63	2.0 ↑	23.07	4.0	-.0000785
3.21	.5 ↑	21.3	2.8	-.0000665
4.78	.75 ↑	20.12	2.0	-.0000563
6.26	1.5 ↑	18.92	1.2	-.0000390
9.4	4.0 ↑	18.35	1.5	-.0000616
12.43	5.25 ↑	17.75	1.5	-.0000735
15.49	7.25 ↑	18.35	2.6	-.0001368

$(\beta^\circ - \alpha^\circ)$  = Angle between normal to chord and resultant vector,  $K_r$

$$K_M \text{ e.g.} = \frac{(\text{Distance from C.G. to vector}) \times K_r}{\text{Chord}}$$

DETERMINATION OF AIRPLANE  $K_y$  :

$$\text{Airplane } K_y = \text{Cellule } K_y + \frac{\text{Tail Load}}{A_w V^2}$$

$$\frac{\text{Tail Load}}{A_w V^2} = K_M \text{ e.g.} \times \frac{A_w V^2}{A_w V^2} \times \frac{C}{d} = K_M \text{ e.g.} \times \frac{59.15}{177.8} = .333 K_M \text{ e.g.}$$

Cellule $\alpha$	$(\text{Cellule } K_y + .333 K_M \text{ e.g.})$	Airplane $K_y$
-9.07		-.000673
-6.04		-.0001933
-4.46		.000075
-2.89		.000338
-1.42		.000595
.16		.0008625
1.63		.0011268
3.21		.001383
4.78		.00164
6.26		.001909
9.4		.002404
12.43		.00284
15.49		.003034

Authority: A.D.M. 1061

Cellule	Airplane $K_A$	Cellule $K_C$	(Cellule $K_C$ .333 $K_A$ .e.g.)	Airplane $K_A$
15.49				.003034
12.43				.00284
9.4				.002404
6.26				.001909
4.78				.00164
3.21				.001383
1.63				.001368
.16				.0008652
-1.42				.000292
-2.89				.000338
-4.46				.000075
-6.04				-.0001933
-9.07				-.000673

DETERMINATION OF AIRPLANE  $K_A$

$$K_A \text{ e.g.} = \frac{\text{Chord}}{(\text{Distance from C.G. to vector}) \times K_T}$$

( - ) Angle between normal to chord and resultant vector,  $K_T$

Cellule	( - )	Inches from I.E. to C.P.	Inches from G.C. to vector	$K_A$ e.g.
15.49	7.25	18.35	2.6	-.0001368
12.43	5.25	17.75	1.2	-.0000732
9.4	4.0	18.35	1.2	-.0000616
6.26	1.2	18.92	1.2	-.0000390
4.78	.75	20.12	2.0	-.0000263
3.21	.5	21.3	2.8	-.0000662
1.63	2.0	23.07	4.0	-.0000792
.16	3.2	26.0	6.25	-.0000946
-1.42	5.25	31.4	10.8	-.0001125
-2.89	8.0	42.2	21.0	-.000137
-4.46	19.0	101.8	70.7	-.000128
-6.04	10.0	72.12	82.8	-.000196
-9.07	1.2	2.21	20.0	-.000208

DETERMINATION OF  $K_A$  e.g.



DETERMINATION OF AIRPLANE  $K_x$  :

$$\text{Airplane } K_x = K_x (\text{cellule}) + \frac{A_e}{A_w} \times .00327 = K_x (\text{cellule}) +$$

$$\frac{5.64}{240.76} \times .00327 = K_x (\text{cellule}) + .0000765$$

$$\text{Cellule } K_x (\text{cellule}) + .0000765 = K_x (\text{airplane})$$

-9.07	.0001915
-6.04	.0001132
-4.46	.0001099
-2.89	.0001118
-1.42	.0001207
.16	.0001319
1.63	.0001515
3.21	.0001698
4.78	.0001991
6.26	.0002345
9.4	.0003155
12.43	.0004075
15.49	.0005155

Authority: A.D.M. 1061, page 16.

DETERMINATION OF VELOCITIES:

$$V = \sqrt{\frac{W}{A}} \sqrt{\frac{\cos \beta}{K_y}} = \sqrt{\frac{2890}{240.76}} \sqrt{\frac{\cos \beta}{K_y}} = 3.46 \sqrt{\frac{\cos \beta}{K_y}}$$

Cellule $\alpha$	Airplane $\frac{L}{D}$	Flight path angle. $\beta = \cot^{-1} \frac{L}{D}$	$\cos \beta$	Airplane $K_y$	$\frac{\cos \beta}{K_y}$	V
-9.07	3.51	15.9	.9617	-.000673	1430	131
-6.04	1.705	30.4	.8625			231.5
-4.46	.683	55.67	.5640			300
-2.89	3.02	18.3	.9494			183
-1.42	4.935	11.45	.9801			140
.16	6.55	8.67	.9886			117.2
1.63	7.43	7.625	.9912			102.8
3.21	8.14	7.0	.9925			92
4.78	8.24	6.92	.9928			85
6.26	8.13	7.01	.9925			79
9.4	7.62	7.48	.9915			70.3
12.43	6.97	8.17	.9922			64.7
15.49	5.87	9.67	.9858			62.4



## DETERMINATION OF TAIL LOADS:

$$\text{Normal Tail Load} = K_{Mcg} \times A_W \times \frac{C}{d} \times V^2 = K_{Mcg} \times V^2 \times 80.25$$

$$\text{Dynamic Tail Load} = \text{Normal Tail Load} \times \left( \frac{V_{\max}}{V} \right)^2$$

Cellule $\alpha^\circ$	$K_{Mcg}$	$V^2$	$\left( \frac{V_{\max}}{V} \right)^2$	Dynamic Tail Load	Normal Tail Load.
-9.07	-.000208	17160	6.35	1820	286.5
-6.04	-.000196	53600	2.035	1710	841.5
-4.46	-.000158	90000	1.21	1380	1140.
-2.89	-.000137	32500	3.25	1197	368.
-1.42	-.000115	19600	5.56	1007	181.
.16	-.0000946	13760	7.94	829	104.4
1.63	-.0000785	10550	10.33	686	66.5
3.21	-.000062	8464	12.87	541.5	42.15
4.78	-.00005	7225	15.1	437	28.95
6.26	-.00004	6241	17.4	348	20.
9.4	-.000047	4942	22.0	409	18.6
12.43	-.000078	4186	26.0	681	26.2
15.49	-.0001368	3894	28.0	1192	42.6

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# DETERMINATION OF TAIL LOADS:

$$\text{Normal Tail Load} = K_{\text{og}} \times A_w \times C \times V = K_{\text{og}} \times V \times 80.22$$

$$\text{Dynamic Tail Load} = \text{Normal Tail Load} \times \frac{V_{\text{max}}}{V}$$

Normal Tail Load	Dynamic Tail Load	$\left(\frac{V_{\text{max}}}{V}\right)$	V	K <sub>og</sub>	Coefficient
28.6	112	38.0	384	-0.001368	15.42
26.2	68	38.0	418	-0.000078	15.42
18.6	40	32.0	424	-0.000047	9.4
20.	34	17.4	641	-0.00004	6.26
28.22	43	12.1	722	-0.00002	4.78
42.12	24.2	12.87	844	-0.00002	3.21
66.2	68	10.22	1020	-0.000078	1.62
104.4	82	7.24	1270	-0.000046	1.6
151.	100	2.26	1260	-0.00012	-1.42
368.	112	3.22	3250	-0.00137	-2.82
1140.	138	1.21	2000	-0.00128	-4.46
841.2	171	2.22	2260	-0.00126	-6.04
286.2	182	6.22	1710	-0.00208	-2.07

Authority A.T.M. 1061.

EFFECT OF C.G. LOCATION ON VALUE OF  $K_{Mcg}$  AT ZERO LIFT :

(1)

C.G. at (40% M.A.C.  
(45% below M.A.C.)

Cellule $\alpha^\circ$	inches from C.G. to Vector	$K_r$	$K_{Mcg}$
-9.07	25.5	-.000615	-.000265
-6.04	90.0	-.000133	-.000202
-4.95	zero lift		
-4.46	63.7	.000132	-.000142
-2.89	15.0	.000387	-.000098
-1.42	5.4	.000631	-.0000576

$K_{Mcg}$  at zero lift = -.000160

(2)

C.G. at (40% M.A.C.  
(On M.A.C.)

Cellule $\alpha^\circ$	inches from C.G. to Vector	$K_r$	$K_{Mcg}$
-9.07	26.1	-.000615	-.0002717
-6.04	94.5	-.000133	-.0002125
-4.46	72.2	.000132	-.000161
-2.89	18.5	.000387	-.000121
-1.42	7.6	.000631	-.0000811

$K_{Mcg}$  at zero lift = -.000175

(3)

C.G. at (25% M.A.C.  
(45% below M.A.C.)

Cellule $\alpha^\circ$	inches from C.G. to Vector	$K_r$	$K_{Mcg}$
-9.07	16.7	-.000615	-.0001738
-6.04	81.3	-.000133	-.000183
-4.46	72.0	.000132	-.0001607
-2.89	23.6	.000387	-.0001543
-1.42	14.1	.000631	-.0001505

$K_{Mcg}$  at zero lift = -.000170

EFFECT OF C.G. LOCATION ON VALUE OF  $K_{\text{MAG}}$  AT ZERO LIFT

(1)

C.G. at (40° M.A.C.)  
(45° below M.A.C.)

Celulose	inches from C.G. to Vector	$K_T$	$K_{\text{MAG}}$
-1.42	5.4	.000631	-.0000276
-2.82	15.0	.000387	-.000028
-4.46	63.7	.000132	-.000142
-4.93	zero lift		
-6.04	90.0	-.000132	-.000028
-9.07	22.5	-.000612	-.000025

$K_{\text{MAG}}$  at zero lift = -.000160

(2)

C.G. at (40° M.A.C.)  
(on M.A.C.)

Celulose	inches from C.G. to Vector	$K_T$	$K_{\text{MAG}}$
-1.42	7.6	.000631	-.0000811
-2.82	18.2	.000387	-.000121
-4.46	72.2	.000132	-.000161
-4.93	94.2	-.000132	-.000122
-6.04	26.1	-.000612	-.000277

$K_{\text{MAG}}$  at zero lift = -.000172

(3)

C.G. at (25° M.A.C.)  
(45° below M.A.C.)

Celulose	inches from C.G. to Vector	$K_T$	$K_{\text{MAG}}$
-1.42	14.1	.000631	-.0001202
-2.82	23.6	.000387	-.0001242
-4.46	72.0	.000132	-.0001607
-4.93	81.3	-.000132	-.000182
-6.04	16.7	-.000612	-.0001738

$K_{\text{MAG}}$  at zero lift = -.000170

EFFECT OF C.G. LOCATION ON VALUE OF  $K_{Mcg}$  AT ZERO LIFT:

(4)

C.G. at (25% M.A.C.  
(On M.A.C.

Cellule $\alpha^\circ$	inches from C.G. to Vector	$K_r$	$K_{Mcg}$
-9.07	17.2	-.000615	-.000179
-6.04	85.7	-.000133	-.0001928
-4.46	80.7	-.000132	-.000180
-2.89	27.4	.000387	-.000178
-1.42	16.6	.000631	-.000177

$K_{Mcg}$  at zero lift -.000185

EFFECT OF C.G. LOCATION ON VALUE OF  $K_{\text{MAG}}$  AT ZERO LIFT.

(4)

C.G. at 25% M.A.C.  
(on M.A.C.)

$K_{\text{MAG}}$	$K_Y$	inches from C.G. to Vector	Ceiling
87.000178	87.000015	17.5	-2.07
88.000180	88.000138	85.7	-6.04
89.000180	89.000138	80.7	-4.46
90.000178	90.000387	87.4	-3.82
91.000177	91.000631	16.6	-1.48

$K_{\text{MAG}}$  at zero lift -- 88.000182



EFFECT OF C.G. LOCATION ON VALUE OF NORMAL TAIL LOAD  
AT ZERO LIFT:

$$\text{Normal tail load} = K_{M_{cg}} \times A_w \times \frac{C}{d} \times V^2$$

- (1) C.G. at (40% M.A.C. )  
 (25% below M.A.C. )

$$d = 177.3 - (.40 - .31) \times 59.15 = 177.3 - 5.3 = 172"$$

$$\text{Normal tail load} = K_{M_{cg}} \times 240.76 \times \frac{59.15}{172.0} \times V^2$$

Cellule $\alpha^\circ$	$(82.8 \times V^2 \times K_{m_{cg}})$	= Normal Tail Load
-9.07	$(82.8 \times 17160 \times -.000265) =$	-376.2
-6.04	$(82.8 \times 53600 \times -.000202) =$	-896.0
-4.46	$(82.8 \times 90000 \times -.000142) =$	-1060.0
-2.89	$(82.8 \times 33500 \times -.000098) =$	-272.0
-1.42	$(82.8 \times 19600 \times -.0000576) =$	-93.5

- (2) C.G. at (40% M.A.C. )  
 (on M.A.C. )

$$d = 172"$$

$$\text{Normal Tail Load} = 82.8 \times V^2 \times K_{M_{cg}}$$

Cellule.

$\alpha^\circ$		
-9.07	$(82.8 \times 17160 \times -.0002717) =$	-386
-6.04	$(82.8 \times 53600 \times -.0002125) =$	-945
-4.46	$(82.8 \times 90000 \times -.000161) =$	-1200
-2.89	$(82.8 \times 33500 \times -.000121) =$	-336
-1.42	$(82.8 \times 19600 \times -.0000811) =$	-131.6

# EFFECT OF C.G. LOCATION ON VALUE OF NORMAL TAIL LOAD AT ZERO LIFT.

$$\text{Normal tail load} = K_{\text{CG}} \times W \times \frac{C}{D} \times V$$

- (1) C.G. at (40% M.A.C.)  
(2) 25% below M.A.C.

$$d = 177.3 - (.40 - .31) \times 29.18 = 177.3 - 2.6 = 174.7$$

$$\text{Normal tail load} = K_{\text{CG}} \times 240.76 \times \frac{29.18}{174.7} \times V$$

$$\text{Normal Tail Load} = (82.8 \times V \times K_{\text{CG}}) \times \text{Cefine}$$

82.8 x 17160 x 8.38	-2.07
82.8 x 23600 x 8.38	-6.04
82.8 x 30000 x 8.38	-4.46
82.8 x 33200 x 8.38	-2.89
82.8 x 19600 x 8.38	-1.42

- (2) C.G. at (40% M.A.C.)  
(3) on M.A.C.

$$d = 175$$

$$\text{Normal Tail Load} = 82.8 \times V \times K_{\text{CG}}$$

Cefine.

82.8 x 17160 x 8.38	-2.07
82.8 x 23600 x 8.38	-6.04
82.8 x 30000 x 8.38	-4.46
82.8 x 33200 x 8.38	-2.89
82.8 x 19600 x 8.38	-1.42

EFFECT OF C.G. LOCATION ON VALUE OF NORMAL TAIL LOAD  
AT ZERO LIFT:

- (3) C.G. at (25% M.A.C. )  
(45% below M.A.C.)

$$d = 177.3 + ( .31 - .25 ) \times 59.15 = 177.4''$$

$$\begin{aligned} \text{Normal tail load} &= K_{M_{cg}} \times V^2 \times 240.76 \times \frac{59.15}{177.40} \\ &= 80.3 \times K_{M_{cg}} \times V^2 \end{aligned}$$

Cellule	$(80.3 \times V^2 \times K_{M_{cg}})$	=	Normal Tail Load
-9.07	$(80.3 \times 17160 \times -.0001738)$	=	-241
-6.04	$(80.3 \times 53600 \times -.000183)$	=	-789
-4.46	$(80.3 \times 90000 \times -.0001607)$	=	-1160
-2.89	$(80.3 \times 33500 \times -.0001543)$	=	-415
-1.42	$(80.3 \times 19600 \times -.0001505)$	=	-237

- (4) C.G. at (25% M.A.C. )  
(on M.A.C. )

$$d = 177.4''$$

Cellule			
-9.07	$(80.3 \times 17160 \times -.000179)$	=	-246
-6.04	$(80.3 \times 53600 \times -.0001928)$	=	-830
-4.46	$(80.3 \times 90000 \times -.000180)$	=	-1303
-2.89	$(80.3 \times 33500 \times -.000178)$	=	-478.5
-1.42	$(80.3 \times 19600 \times -.000177)$	=	-278

REMARKS OF C.C. LOCATION ON VALUE OF NORMAL TAIL LOAD  
AT ZERO LIFT.

(3) C.C. at (25% M.A.C.)  
(45% below M.A.C.)

$$d = 177.3 + (.31 - .25) \times 29.15 = 177.4$$

$$\text{Normal tail load} = K_{\text{No}} \times V \times 240.76 \times 29.15$$

$$= 80.3 \times K_{\text{No}} \times V$$

$$177.40$$

Normal Tail Load		(80.3 x V x K <sub>No</sub> )	Coefficient
-1.45	=	(80.3 x 19600 x -.0001202)	-1.45
-2.89	=	(80.3 x 33200 x -.0001243)	-2.89
-4.46	=	(80.3 x 50000 x -.0001607)	-4.46
-6.04	=	(80.3 x 66600 x -.000183)	-6.04
-9.07	=	(80.3 x 17160 x -.0001738)	-9.07

(4) C.C. at (25% M.A.C.)  
(on M.A.C.)

$$d = 177.4$$

		(80.3 x V x K <sub>No</sub> )	Coefficient
-1.45	=	(80.3 x 19600 x -.0001202)	-1.45
-2.89	=	(80.3 x 33200 x -.0001243)	-2.89
-4.46	=	(80.3 x 50000 x -.0001607)	-4.46
-6.04	=	(80.3 x 66600 x -.000183)	-6.04
-9.07	=	(80.3 x 17160 x -.0001738)	-9.07

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